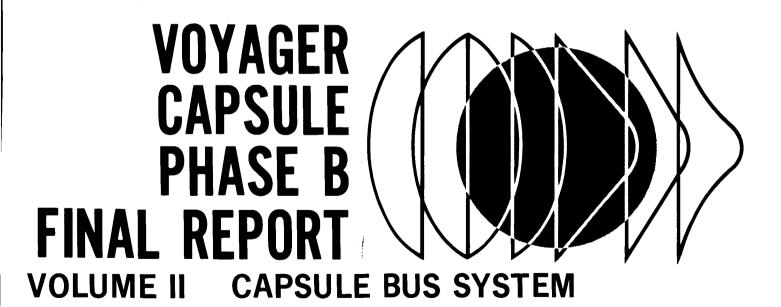
PART B2 ALTERNATIVES, ANALYSES, SELECTION



PREPARED FOR:
CALIFORNIA INSTITUTE OF TECHNOLOGY
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REPORT ORGANIZATION

VOYAGER PHASE B FINAL REPORT

The results of the Phase B Voyager Flight Capsule study are organized into several volumes. These are:

Volume I Summary

Volume II Capsule Bus System

Volume III Surface Laboratory System

Volume IV Entry Science Package

Volume V System Interfaces

Volume VI Implementation

This volume, Volume II, describes the McDonnell Douglas preferred design for the Capsule Bus System. It is arranged in 5 parts, A through E, and bound in 11 separate documents, as noted below.

Part A	Preferred Design Concept	2 documents, Parts A_1 and A_2
Part B	Alternatives, Analyses, Selection	5 documents, Parts B ₁ ,
		B_2 , B_3 , B_4 and B_5
Part C	Subsystem Functional Descriptions	2 documents, Parts ${ t C}_1$
		and ${\rm C}_2$
Part D	Operational Support Equipment	1 document
Part E	Reliability	1 document

In order to assist the reader in finding specific material relating to the Capsule Bus System, Figure 1 cross indexes broadly selected subject matter, at the system and subsystem level, through all volumes.

VOLUME II CROSS REFERENCE INDEX

_		PART A	PARTR	O TOVO	C + 0 4 0	DADT E
/	VOLUME II PARTS	DESCRIPTION OF	AL TERNATIVES	DETAIL ED DE.	OPERATIONAL SUP.	RELIABILITY CON-
_		PREFERRED SYS.	ANAL YSIS AND	SCRIPTION OF	PORT EQUIPMENT	STRAINTS, ANALY-
	_	TEM OBJECTIVES,	SELECTION -	SUBSYSTEM	 SYSTEM, SUBSYS- 	SIS, RESULTS, PRO-
		MISSION, DESIGN,	METHODS TRADE	FUNCTIONS	TEM, LAUNCH COM-	GRAM TESTING,
	/	SUBSYSTEMS,	SIUDIES, OPTIMI-		PLEX, MISSION,	CONTROL
SYSTEM/	SYSTEM/SUBSYSTEM	OPERATIONS, SUP- PORTING FUNC. TIONS	ZATION STUDIES RESULTS		MARE WARE	
CAP	CAPSULE BUS SYSTEM					
	Objectives	1.1-Summary	2-Analysis	N/A	1-General	1-Constraints
Mission	Profile	1.2-Summary	2—Analysis 2.4—Selection	N/A	N/A	3.1.1—Analysis
	Operations	4-Description by Phase	2.3-Analysis 2.3.7-Landing Site	N/A	4.4-LCE Description 3-Estimates	3-Estimates
			Select		<u> </u>	
	General	2-Criteria Summary	1-Study Approach	N/A	3.2-Concept	4-Program Require-
·		3. 1—Configuration	3-Functional Ke- quirements		3.3—Summary 6.1—AHSE	5-Component Reli-
						delitik
Design	Standardization/Growth 2.5—Summary 11—Future	2.5—Summary 11—Future	(See Specific Subsystem Below)	۷ ۲	4.3.8-STC) 4.4.8-LCE Srowth	۷ ۷
					4.5.8-MDE J	
	Weight	3.1.2.4—Summary 5—Breakdown	(See Specific Subsystem Below)	N/A	N/A	2.3.2-Reliability vs Weight
Interfaces	Interfaces (See Also Vol. V)	3.1-Summary 9.0-Operational	(See Volume V)	N/A	4.2.1,4.3.5,4.4.5, 4.5.5	N/A
Implemento	Implementation (See Also Vol. VI)	10-Schedule 8.11-OSE	(See Volume VI)	N/A	(See Specific Subsystem Below)	N/A
Planetary Quarantine	Quarantine	7-General	(See Volume VI, C,7 Sterilization Plan)	N/A	None Required	N/A
0.S.E. (See	O.S.E. (See Also Part D)	8-General	(See D2.5-Selection	(See D5-Subsystem	Complete OSE	(See D4.3.6-STC
		(See Also-D1, D2, D3 D4)	Criteria, Dy-Analy- sis DIO - Alterna-	ment See Also DA	Description	D4.4.6-LCE
		(t	tives)	D6, D7)		
	H 0 20					
	SUBSTS1 EMS					
Sterilization Canister	ın Canister	3.2.1.1-Description 3.1.2-Summary	5.1-Analysis	<u> </u>	6.1.5.2,6.1.5.3,	(See Part C Sections 1, 1, 1, 7
		•			6.1.5.8-AHSE 6.2.15-Servicing	1.1.2.7
Adapter		3.2.1.2-Description 3.1.2-Summary	5.2-Analysis	1.2	None Required	(See Part C, 1.2.7)

Figure 1 --/

Aerosneii	3.2.1.3—Description	4.1-Contiguration Selection	1.3	None Required	(See Part C, 1.3.7)
		4.6—Separation 5.3.1—Structure 5.3.2—Heat Shield			
Lander	3.2.1.4—Description 3.1.2—Summary	4.2-Configuration Selection 5.4-Analysis	1.4	6.1.5.9-Fixture	(See Part C, 1.4.7)
Telecommunications	3.2.2.1-Description	4.9-In-Flight Moni- toring 5.5-Analysis	2-Telemetry 3-Radio 4-Antenna 5-Data Storage 6-Command	4.3.9.1-STC Console (See Part C, Sections 5.7-Test Set 2.1.7, 2.2.7, 2.3.7, 3.1.7, 3.2.7, 4.5, 5.1.7, 5.2.7, 6.1.7, 6.2.7)	(See Part C, Sections 2.1.7, 2.2.7, 2.3.7, 3.1.7, 3.2.7, 4.5, 5.1.7, 5.2.7, 6.1.7, 6.2.7)
Power	3.2.2.2-Description	5.6-Analysis	7	4.3.9.1-STC Console 5.3-Test Set	(See Part C, Sect. 7.7)
Sequencing and Timing	3.2.2.3—Description	5.7—Analysis	8	4.3.9.1-STC Console 5.4-Test Set	(See Part C Sections 8.1.7, 8.2.7)
Guidance and Control	3.2.2.4—Description	4.7-De-orbit Atti- tube Regm*t. 5.8-Analysis	6	4.3.9.1_STC Console (See Part C, 9.7) 5.5_Test Set	(See Part C, 9.7)
Radar	3.2.2.5—Description	5.9—Analysis	10	4.3.9.1-STC Console (See Part C, Sections 5.6-Test Set	(See Part C, Sections 10.1.7, 10.2.7)
Aerodynamic Decelerator	3.2.3—Description	4.4—Selection 5.10—Analysis	11	None Required	(See Part C, 11.7)
Pyrotechnics	3.2.4-Description	5.11-Analysis	12	4.3.9.1-STC Console 5.9-Test Set	(See Part C, 12.7.1)
Thermal Control	3.2.5-Description	5.12-Analysis	13	4.3.9.1-STC Console 5.8-Test Set	(See Part C, 13.6)
De-orbit Propulsion	3.2.6.1-Description	5.13.1-Analysis	14	5.10-Test Set 6.1.5.10-AHSE	(See Part B, 5.13.4.5) (See Part C, 14.7)
Reaction Control	3.2.6.2—Description	5, 13, 2 - Analysis	15		(See Part B, 5.13.4.5) (See Part C, 15.7)
Terminal Propulsion	3.2.6.3—Description	4.3—Configuration Selection 4.5—Terminal Deceleration 5.13.3—Analysis	16	4.3.9.1-STC Console 5.10-Test Set 6.2-Servicing	(See Part B, 5.13.4.5) (See Part C, 16.7)
Packaging and Cabling	3.2.7 - Description	5.14—Analysis	71	4.3.9.8—Special Purpose Complex Equipment	Not Required

Note: Parentheses Refer Reader to Volumes/Parts Outside of the Respective Notation

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SECTION 4

MAJOR TRADE STUDIES AND SYSTEMS ANALYSES

The major trade studies and analyses of the Capsule Bus System described in this section are those affecting the overall design to an important degree or involving several subsystems and technologies. The starting point for each trade study is the baseline document shown in Figure 3-1. In this document:

- a. Functions to be performed are identified.
- b. Alternative implementations are identified.
- c. A preliminary selection (the baseline) is made.
- d. Major trade studies are identified.
- e. Selections are modified as the analysis proceeds.

This technique assures the use of a consistent system for all studies at a given time.

The criteria used to evaluate the candidate approaches were probability of mission success, system performance, development risk, versatility, and cost, with weighting as discussed in Section 1.

Figure 4-1 shows the interdependence of these trade studies. An iteration resulted in some of the studies as the baseline was changed, since several studies were conducted concurrently. For instance, the Aeroshell/lander separation study was conducted when a Ballute was the baseline deployable aerodynamic decelerator. When the preferred design included a parachute, the separation study was reevaluated to determine its sensitivity to the change. The conclusions were found to be still valid. Similarly, the lander study was conducted assuming a six engine terminal propulsion configuration. The selection of a four engine configuration occurred almost concidentally with a new set of surface environment constraints so that this study was conducted in two distinct phases.

MAJOR TRADE STUDY INTERDEPENDENCE

IDENTIFICATION	EFFECT OF OTHER STUDIES FFFE	EFFECT ON OTHER STUDIES
4.1 Aeroshell Configuration Selection	Aerodynamic decelerator stowage and deployment suitability — Lander shape compatibility.	Lander/Surface Lab, envelope, c.g., inertia requirements. Aeroshell dragestablishes initial conditions for terminal decelerator.
4.2 Lander Configuration Selection	Installation interfaces with terminal decelerator configuration selection.	Lander mass affects size of aerodynamic decelerator needed for separation. Establishes terminal decelerator end conditions.
4.3 Terminal Propulsion Selection	Aerodynamic decelerator establishes the initial conditions of velocity and altitude.	Establishes equilibrium conditions (velocity and altitude) for aerodynamic decelerator.
4.4 Deployable Aerodynamic Decelerator Selection	Aeroshell configuration determines need for and size of aerodynamic decelerator. Separation technique determines sizing requirements.	Terminal propulsion requirements are set by aerodynamic decelerator selection and sizing.
4.5 Terminal Decelerator Selection	Candidates are screened in terminal propulsion and aerodynamic decelerator selection studies.	Establishes Lander impact áttenuator requirements.
4.6 Aeroshell/ Lander Separation Selection	Decision to use a deployable aerodynamic decelerator allows another separation technique to be studied — differential drag.	Selection determines need for and/or size of aerodynamic decelerator.
4.7 Orbital Descent Attitude Requirements	Solar angle limits set by cone angle of Aeroshell.	

- 4.1 AEROSHELL CONFIGURATION SELECTION The Aeroshell configuration selected for the VOYAGER mission must conform to various mission objectives. These include:
 - a. High probability of mission success
 - b. Maximum scientific payload
 - c. Design flexibility
 - d. Minimum development time and cost

The Aeroshell's primary requirement is a large deceleration capability to meet these objectives in the sparse Martian atmosphere. Three families of shapes can satisfy this requirement:

- a. Spherical segment
- b. Tension shell
- c. Large angle sphere-cone

These shapes were evaluated with respect to the mission objectives on the basis of their aerothermodynamic, structural - mechanical, and mission entry science characteristics. From this evaluation a 120° sphere-cone of .5 nose-to-base radius bluntness ratio, a corner radius-to-base radius ratio of .005, and a diameter of 19 ft was selected.

- 4.1.1 <u>Aeroshell Characteristic Requirements</u> To meet the mission objective the Aeroshell must have suitable characteristics in three areas: aerothermodynamic, structural and mechanical, and mission entry science.
- 4.1.1.1 <u>Aerothermodynamic Characteristics</u> The prime requirement of the external shape is to decelerate the payload from entry velocity to the point of terminal decelerator initiation. To maximize the payload weight the shape should have a large drag effectiveness, C_nA .

During atmospheric entry the Capsule Bus should be stable to:

- a. <u>Maximize Deceleration</u> Drag force is largest at zero angle-of-attack for the blunt shapes considered.
- b. <u>Minimize Heat Protection and Structural Weights</u> Loads and aerodynamic heating associated with non-zero angle-of-attack conditions require additional structure and heat protection.
- c. Simplify the Telecommunication Subsystem Oscillation of the capsule will

- require larger antenna beam-widths and an increase in power.
- d. <u>Facilitate the Science Experimentation</u> Radar, television, and atmospheric sensors will operate more effectively if the vehicle does not oscillate.
- e. <u>Facilitate Auxiliary Aerodynamic Decelerator Deployment</u> Deployment and inflation problems are minimized if the vehicle does not experience severe oscillations.

The configuration should have well defined aerodynamic and aerothermodynamic characteristics and require minimum heat protection weight for high payload capability and improved probability of mission success.

4.1.1.2 <u>Structural and Mechanical Characteristics</u> - The Aeroshell structure must accommodate the loads and heating experienced during deceleration. The shape should be one which can be easily fabricated. Shapes having small areas of compound curvature and which can be made in segments and riveted or spot welded together are desirable.

The volume and shape of the vehicle are constrained by the launch configuration dynamic envelope. Configurations having their maximum diameter near the base are limited to 19 ft by the launch fairing and allowances for the Sterilization Canister.

Large depth and volume is desirable for payload accommodation without compromising aerodynamic stability by rearward movement of the center of gravity. A large maximum diameter insures a high drag effectiveness, $C_{\mathsf{D}}A$.

Provisions for access to the internal equipment during assembly and checkout must be provided. Auxiliary aerodynamic decelerator stowage and deployment and easy Aeroshell/lander separation are necessary characteristics of the shape.

4.1.1.3 Entry Science Mission Characteristics - Entry science measurements and capsule-spacecraft communications during descent are required as part of the mission. Atmospheric properties to be measured during descent include pressure, density, temperature, and gas composition. The Aeroshell design must permit installation of the sensors in locations permitting acquisition of reliable data.

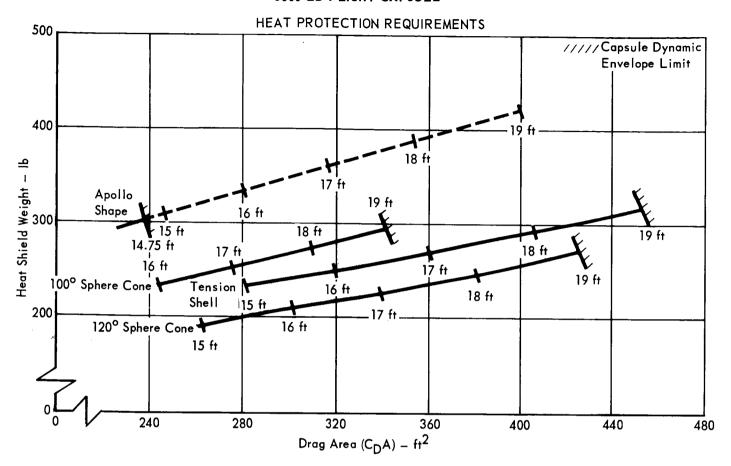
Uninterrupted capsule-to-spacecraft communication during entry is desired for transmission of the entry science measurements and capsule engineering performance data. This requires that communication black-out due to ionized gases in the shock layer surrounding the vehicle be minimized.

- 4.1.2 <u>Candidates</u> Three families of shapes were selected as suitable candidates for the VOYAGER mission: spherical segments, tension shells, and large angle sphere-cones. A characteristic shape was selected from each family for trade study purposes. A modified Apollo shape was chosen from the spherical segment family. A tension shell with nose-to-base radius ratio of .3 and afterbody tangent to the nose at a 46° angle was chosen as the best member of this family based on load, stress, and aerodynamic considerations. A 120° sphere-cone of .5 nose-to-base radius ratio was chosen from the sphere-cone family.
- 4.1.3 <u>Evaluation for Mission Objectives</u> The following presents the data used to evaluate each configuration with respect to the mission objectives for the purpose of selecting the best family of aerodynamic shapes.
- 4.1.3.1 Aerothermodynamic Evaluation For trade-off purposes in estimating heat protection and structural weight, it is sufficient to use the nearly constant hypersonic drag coefficient since this flight regime provides most of the deceleration. All three shapes provide a C_D of 1.4 to 1.6 but the launch vehicle dynamic envelope limits the diameter of the Apollo shape most severely so it has the lowest drag effectiveness (C_DA). The agreement between theory and experimental aerodynamic data is much better for the Apollo and 120° sphere-cone than for the tension shell. Therefore, the effects of changes in shape due to cone angle, nose bluntness, and corner rounding can be predicted with more confidence for these shapes than for the tension shell. In addition, the absence of flow separation on the forebody make these shapes relatively insensitive to Reynolds number and angle of attack.

In contrast to the 120° sphere-cone, the tension shell drag and stability is known to be very dependent on angle-of-attack, Reynolds number, wall temperature, and Mach number. This makes the shape sensitive to atmospheric composition. The tension shell family is also very sensitive to nose bluntness and cone angle, which makes it difficult to predict or interpret the aerodynamic characteristics with confidence and will require substantially more wind tunnel testing than the other candidate shapes.

To evaluate heat protection requirements, a calculation was made of the heat shield weight required for the S-6 ablator in the VM-3 atmosphere, for an entry velocity (V_e) of 13,000 ft/sec, entry angle (γ_e) of -10.4°, and an M/C_DA for each family corresponding to a 5,000 lb. Flight Capsule. Figure 4.1-1 shows the heat shield weight for the various configurations.

WEIGHT COMPARISON 5000 LB FLIGHT CAPSULE



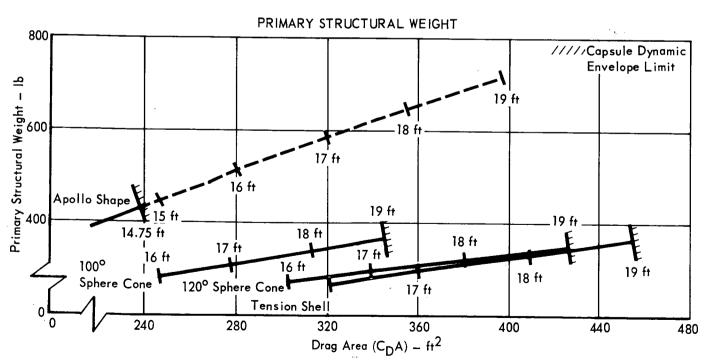


Figure 4.1-1

4-6

The Apollo shape requires large amounts of ablation material on the afterbody. The 120° sphere-cone and tension shell require ablation material only on the forebody. The sphere-cone and tension shell are superior to the Apollo shape with respect to minimum surface area and heat protection weight.

4.1.3.2 Structural and Mechanical Evaluation - A comparison of the structural weight for a 5000-1b Capsule Bus is shown in Figure 4.1-1. The curves were obtained for entry conditions of $M/C_DA = 0.3 \text{ slugs/ft}^2$, $V_e = 15,000 \text{ ft/sec}$, and $\gamma_e = -20^\circ$ in the VM-8 atmosphere. These are the worst conditions from the standpoint of entry aerodynamic loads. The weight includes only the structural shell, aft closure, and aft closure separation joint. It does not include the weight for equipment and payload support, or the weight for the ablative heat protection. A 19 foot, 120° sphere-cone requires slightly less structural weight than a 19 foot tension shell. Each requires over 100 1b less than the 14.75 foot Apollo. The Apollo is clearly the least desirable choice.

Fabrication, Packaging, and Volume - Of the three families considered, the spherecone is the easiest to fabricate because it has a small area of compound curvature and is readily made in segments which can be riveted or spot welded together. This fabrication technique is similar to well proven designs. The tension shell poses the greatest fabrication problems. Close tolerances are required because out-of-contour variations have a significant effect on the stress in the skin. The Apollo shape provides the greatest volume and depth but its $\mathbf{C}_{D}\mathbf{A}$ is limited by the launch configuration dynamic envelope. An increase in base area requires a new shape definition and thus additional aerodynamic testing. The small base area makes decelerator installation and deployment more difficult than for the spherecone and tension shell.

Both the sphere-cone and tension shell have sufficient volume and depth.

4.1.3.3 Entry Science Mission Characteristics Evaluation - Our studies have shown that ablative products make the most significant contributions to communications blackout. All three shapes will experience communication blackout but the effects can be calculated with more certainty on the sphere-cone and Apollo than on the tension shell because of the less complicated flow fields. The uncertainty in blackout time prediction will be greater for the tension shell than the other shapes.

The complex flow field of the tension shell not only affects the blackout predictions but will adversely affect sensor location and data interpretation

for the atmospheric property reconstruction. The relatively simple flow fields about the sphere-cone and Apollo are advantageous for this phase of the VOYAGER mission.

4.1.4 Comparison and Family Selection - Figure 4.1-2 summarizes the evaluation of the candidates characteristics with respect to the Aeroshell requirements to meet the mission objectives. Tension shells are the least desirable family for the VOYAGER mission. Though possessing a large $C_{\overline{D}}A$, the tension shell stability and flow field characteristics are the most undesirable of those evaluated. The close tolerances required increase manufacturing cost and degrade mission success probability.

Though they have similar aerodynamic qualities, the sphere-cone is superior to the Apollo in all structural and mechanical characteristics and heat protection capabilities. In addition, the sphere-cone is best able to accommodate payload growth and design modifications for future missions.

With these considerations, we feel that the large angle sphere-cones provide the best family of shapes for the VOYAGER mission; hence, we directed our efforts to find the optimum sphere-cone shape.

- 4.1.5 Optimum Sphere-Cone Selection Selection of the optimum spherically blunted cone shape is based on considerations of drag, stability, heat protection, and structural characteristics which best satisfy the mission objectives. Spherecones with included angles of 100, 120, and 140 degrees and bluntness ratios from 0 to 1 were considered. Configurations with cone angles less than 100° were not considered because their drag is insufficient to provide the deceleration capability required. Configurations with cone angles greater than 140° were not considered because it is believed they do not possess sufficient stability and do not now have the same degree of wind tunnel test background as the more pointed shapes.
- 4.1.5.1 <u>Aerodynamic Characteristics</u> Aerodynamic characteristics were estimated from Newtonian theory and compared with experimental data. Figure 4.1-3 presents Newtonian calculations for each cone angle as a function of bluntness ratio. Comparison with experimental data is made for the 100° and 120° shapes. Experimental data on the 140° shape is unavailable to us at this time.

The Newtonian calculations show that the aerodynamic characteristics depend very little on bluntness ratio except for the 100° sphere-cones with large bluntness ratio.

				AERO	THERMODYN	IAMIC	
		GEOMETRY	DECELE	RATION		THERMAL	EN SCI
			CD	C _D A	STABILITY	PROTECTION	MI
Modified Apollo Shape	33° 9.2 ft	Surface Area = 495 ft ² Volume = 795 ft ³	1.40	239 ft ²	Adequate static stability margins	Largest heat shield weight requirements	Suitab entry ments
100° Sphere Cone	6.5 ft 50°	$R_N/R_B = .5$ Surface Area = 357 ft ² Volume = 740 ft ³	1.22	346 ft ²		Simple shape for heat trans- fer predic- tion	
120° Sphere Cone	4.8 ft 60°	$R_N/R_B = .5$ Surface Area = 326 ft ² Volume = 521 ft ³	1.50	426 ft ²		Lowest heat shield weight re- quirements	
Tension Shell	19 ft 4.5 ft	$R_N/R_B = .3$ Surface Area = 316 ft ² Volume = 500 ft ³		454 ft ² Largest Aerodynan Uncertain	I	Large heat shield weight re- quirements	Lea suit for mea

Figure 4.1~2

CONFIGURATION EVALUATION

TRY	STRUCTURAL		AL AND MECH	IANICAL			MISSION OBJ	ECTIVES	
	STRUCTURAL DESIGN	FABRICATION	PACKAGING	ACCESSIBILITY	SIZE	MISSION SUCCESS	DEVELOLMENT	PAYLOAD	FLEXIBILITY
e for easure-	Highest structural weight	Large compound curvature area, large aft erbody	Large volume, good depth. Difficult to store and deploy auxilary decelerator.	Poor access from small end. Access to front re- quires panels in heat shield.	Apollo shape must be modified to achieve diameter	Simple shape, with high con- fidence from Apollo devel- ment.	Reduced development because of similarity to other flight programs	Inter- mediate payload weight	Good potential capability to accommodate payload growth and design modifications due to new environmental data
it ible ntry	Low structural compound curvature area. Forebody can be fabricated in segments Least Out-of-contour structural variations will		Large volume. Adequate volume growth possible with afterbody	Easily accesible from base. Access to front re- quires nose cap removal of access panels in heat shield.	Maximum base dia- meter	Simple shape, experimental data and theory correlate well Most complex shape, requires advanced technology.	cause of similarity to other flight programs.	High payload weight	Most sensitive to flight environment

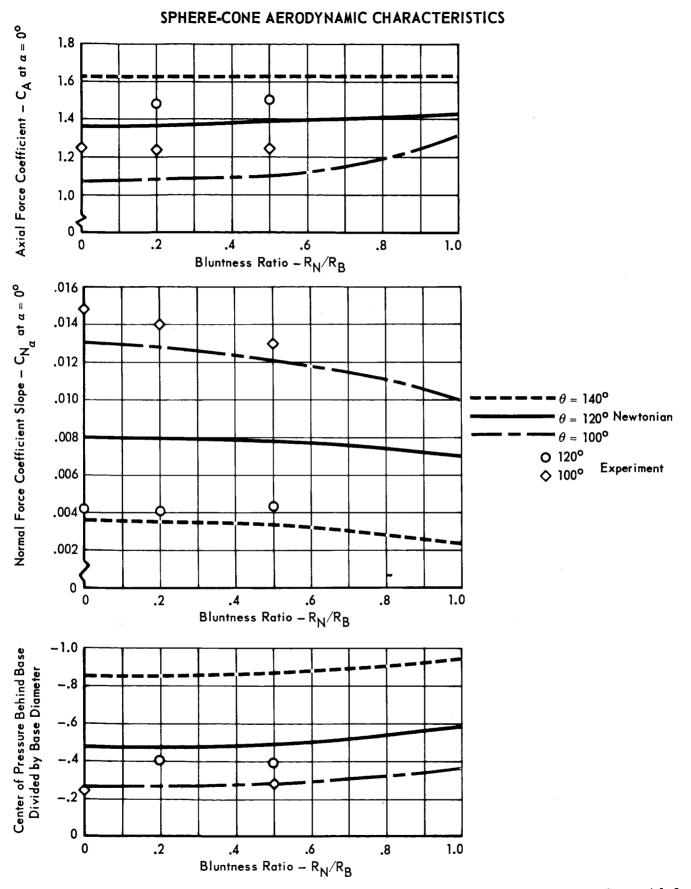


Figure 4.1-3 4-10

To verify this conclusion and to provide a better definition of our baseline configuration, we conducted a wind tunnel test in the McDonnell Polysonic Wind Tunnel (PSWT) on VOYAGER shapes in the Mach number range of 0.7 to 4.87 and Reynolds number range of 7×10^6 to 14×10^6 based on model diameter.

Pictures and shadowgraphs of the three models tested are shown in Figure 4.1-4. All three models were 120° cones, but had variations in the nose radius-to-base radius ratio, $\frac{R_D}{R_D}$, and the corner radius-to-base radius, $\frac{R_C}{R_D}$. The model identification code and geometrical ratios are:

<u>Model</u>	$\frac{R_n}{R_b}$	$\frac{R_{C}}{R_{b}}$
В1	•2	0
B ₂	•5	0
В3	•5	.05

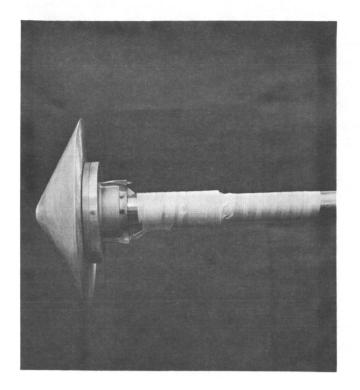
Figure 4.1-5 summarizes the results of this test. Axial force coefficient, normal force coefficient slope, and moment coefficient slope per degree, all evaluated at zero degree angle of attack, are presented as a function of Mach number. The results confirm that nose bluntness and limited corner rounding have little effect on the aerodynamic characteristics of the 120° configuration.

4.1.5.2 Structural and Mechanical Evaluation - In Figure 4.1-1, data was also presented for the heat protection and structural weights required for a 100° sphere-cone of .5 bluntness. This shows that for the same weight capsule the 120° sphere-cone of identical bluntness requires less weight for these functions, so a greater payload can be carried.

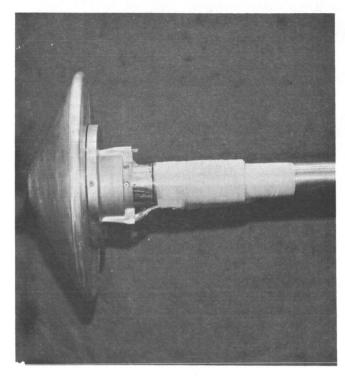
The relationship between entry weight and size is graphically demonstrated by the specific example of the 120° sphere-cone (Figure 4.1-6). The following ground rules were used: de-orbit velocity increment of 950 ft/sec, and auxiliary aero-dynamic decelerator to provide deceleration from about 23,000 ft, attitude control for pre-entry flight, attitude rate damping for atmospheric flight until terminal propulsion ignition, attitude control during terminal descent, and a soft landing subsystem.

Based on these assumptions, a weight breakdown was determined as a function of base diameter for a 5000 lb capsule. The Surface Laboratory weight is maximized for about a 17 foot diameter Aeroshell. The loss in Surface Laboratory weight with a 19 ft diameter is about 26 lbs.

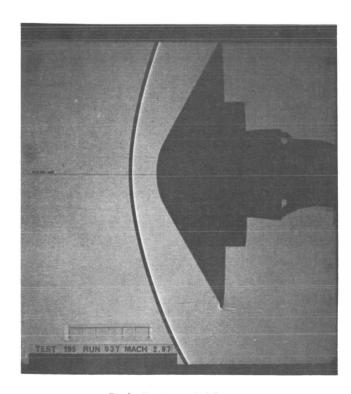
VOYAGER CONFIGURATIONS TESTED IN MCDONNELL POLYSONIC WIND TUNNEL



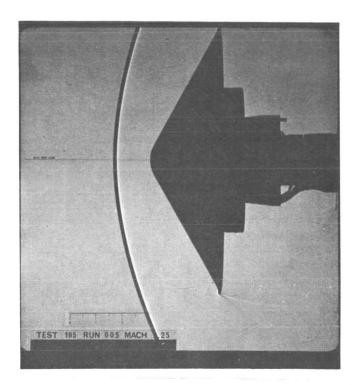
Body B₁



Body B₃



Body $B_1 M_{\infty} = 2.25$



Body B₂ $M_{\infty} = 2.97$

Figure 4.1-4

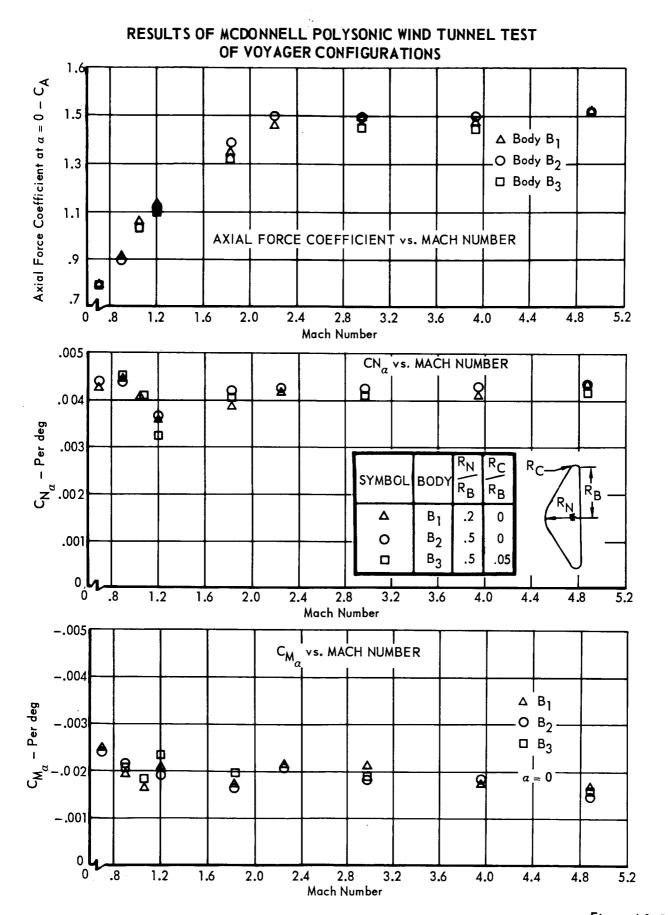
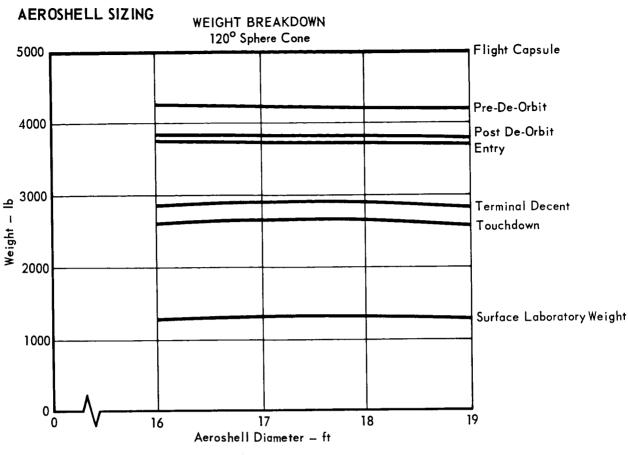


Figure 4.1-5 4-13



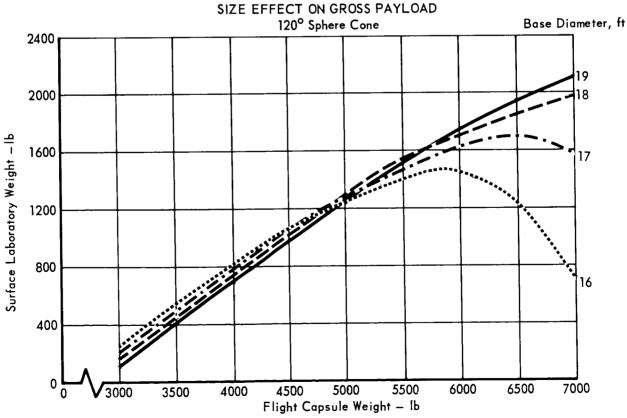


Figure 4.1-6

The effect of capsule weight on optimum size is also illustrated in Figure 4.1-6. Here the Surface Laboratory weight is shown as a function of capsule weight. From the figure, standardizing the diameter to 17 ft based on a 5000 lb capsule would penalize the payload for any possible future increase in capsule weight. The advantages of the 19 foot shape in providing a larger volume, greater depth, and a higher drag effectiveness at the modest penalty of only 26 lbs of payload far outweighs this small disadvantage.

4.1.6 Evaluation - The 140° shape was eliminated from consideration by stability and packaging problems. Though no experimental data is available to us at this time on the 140° shape, conclusions about the characteristics were inferred from the trends with cone angle exhibited by the 100° and 120° data. These trends infer that the 140° shapes experiences higher drag than the 100° and 120° shapes but their static stability is limited by the less than predicted rearward movement of the center of pressure location and the near-zero normal force coefficient slope. In addition, the 140° shapes are volume and depth limited. The volume and depth of a 140° sphere-cone of .5 bluntness and 19 foot diameter is 335 cubic feet and 3.4 feet respectively. This compares with 521 cubic feet and 4.8 feet for a 120° cone of .5 bluntness. These factors require that a significant portion of the lander must be packaged in an afterbody with a further loss in stability due to the rearward movement of the center of gravity.

These factors eliminated the 140° shapes from further consideration. Comparison can then be made between the 120° and 100° sphere-cones. A nominal bluntness of .5 was used for comparison purposes.

Though the 100° sphere-cone has a larger volume and depth than the 120° sphere-cone, the latter has the larger drag effectiveness and requires less heat protection and structural weight. This allows a larger payload capability and, coupled with adequate stability, makes the 120° sphere-cone more suited for the VOYAGER mission than the 100° sphere-cone.

4.1.7 Recommended Configuration - From the results of this study we recommend a 120° sphere-cone of .5 nose-to-base radius ratio, a corner radius-to-base radius ratio of .005, and a base diameter of 19 feet. As bluntness is increased heat shield weight decreases, allowing a larger payload weight, but the volume and depth decrease and the area of compound curvature to be fabricated increase. Together these factors indicate a preferred nose bluntness of one-half the base radius - $(R_{\underline{n}} = .5)$.

4-15

The nominal corner rounding of .005 $R_{\mbox{\footnotesize{B}}}$ is required for heating and structural purposes and the results of our wind tunnel test indicate no significant difference in aerodynamic characteristics from those of the sharp corner shape.

- 4.2 LANDER CONFIGURATION SELECTION The study to select a preferred lander configuration has been a continuing effort for over two years, starting with the hard lander concepts of the early VOYAGER Program. The findings of these early studies were useful in developing candidates for the subsequent soft lander studies. Two distinct sets of design constraints were used during the study. The early soft lander studies were evaluated against the constraints supplied with the VOYAGER Capsule Phase B RFP (Reference 4.2-1). The Phase B contract studies were conducted under a revised set of constraints (Reference 4.2-2). The major differences affecting the lander configuration selection were:
 - a. Increased surface slope angle to local horizontal from + 30° to + 34°.
 - b. Increased abrupt change in slope angles, including ridge and trough formations and conical hills and depressions from 30° total to 68° total.
 - c. Increase in Flight Capsule weight from 5000 lbs. to 7000 lbs. for later missions.
 - d. Mobile Surface Laboratory of at least 1500 lbs. for later missions.
 - e. Emphasis on a standardized Capsule Bus.

Obviously, some of the lander configurations studies under the earlier constraints would not have been considered under the later set, but some of the features found to be desirable were incorporated into new candidate configurations for the Phase B studies. The evolution of the preferred design is shown in Figure 4.2-1. This section is organized to present:

- a. First, the preferred design (which is called the Uni-Disc), its advantages and disadvantages, and a summary of the drop test program conducted with a 1/10 scale model.
- b. Second, the Phase B study which led to the preferred configuration.
- c. Finally, to complete the historical record, the study conducted under the earlier constraints.
- 4.2.1 <u>Preferred Configuration</u> The preferred lander configuration, the Uni-Disc, is a large, flat circular disc shape having a very low silhouette. It thus has a low c.g. and, consequently, is highly stable. The general arrangement is shown in Figure 4.2-2. The design consists of a structural base platform to which are mounted the Surface Laboratory, the terminal descent propulsion subsystem, landing radar antenna and supporting electronics, the landed portion of the Entry Science Package equipment, and an impact attenuation assembly.

The impact attenuation assembly absorbs the landing impact energy while not allowing the Surface Laboratory to tumble or to feel inertia loads in excess of 14 g by providing a low c.g. and a wide footprint. The energy is absorbed by an

LANDER CONFIGURATION EVOLUTION

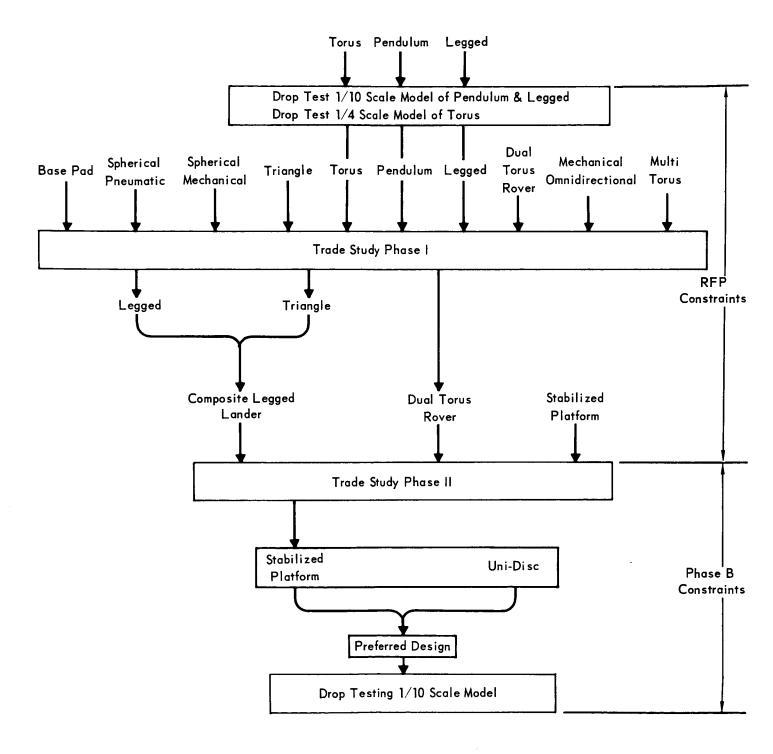
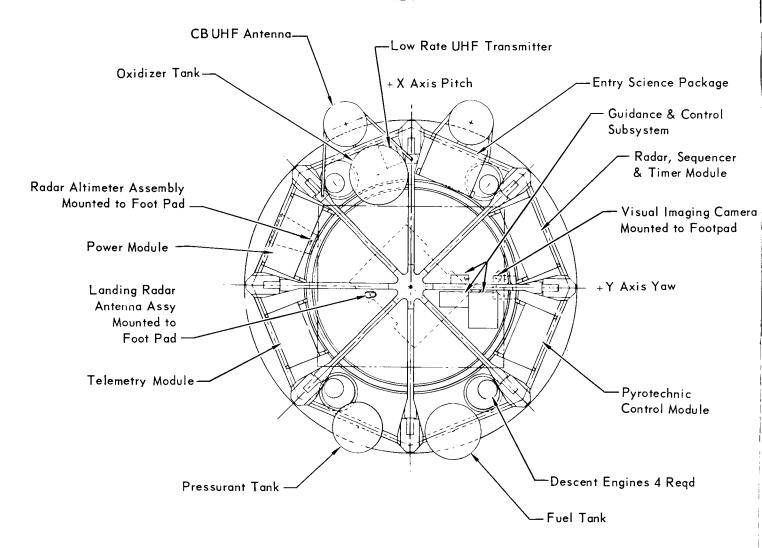
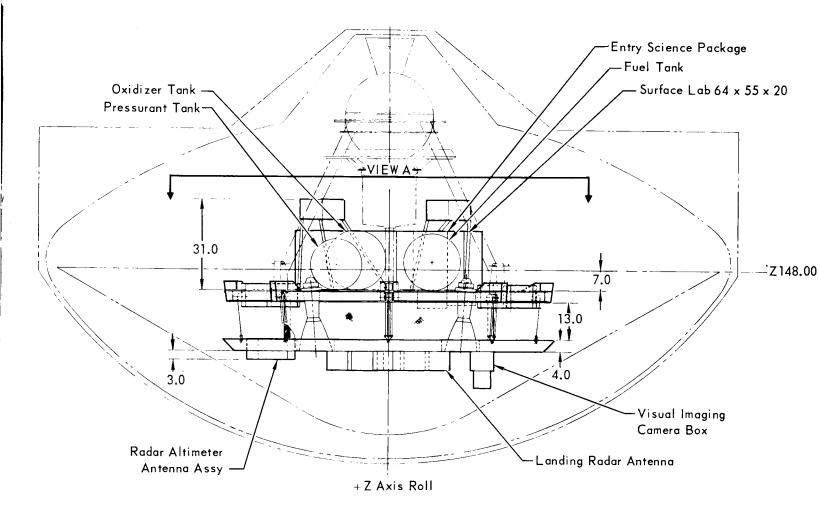


Figure 4.2-1

LANDER GENERAL ARRANGEMENT

VIEW A





CAPSULE LANDER

LANDER GENERAL ARRANGEMENT (Continued)

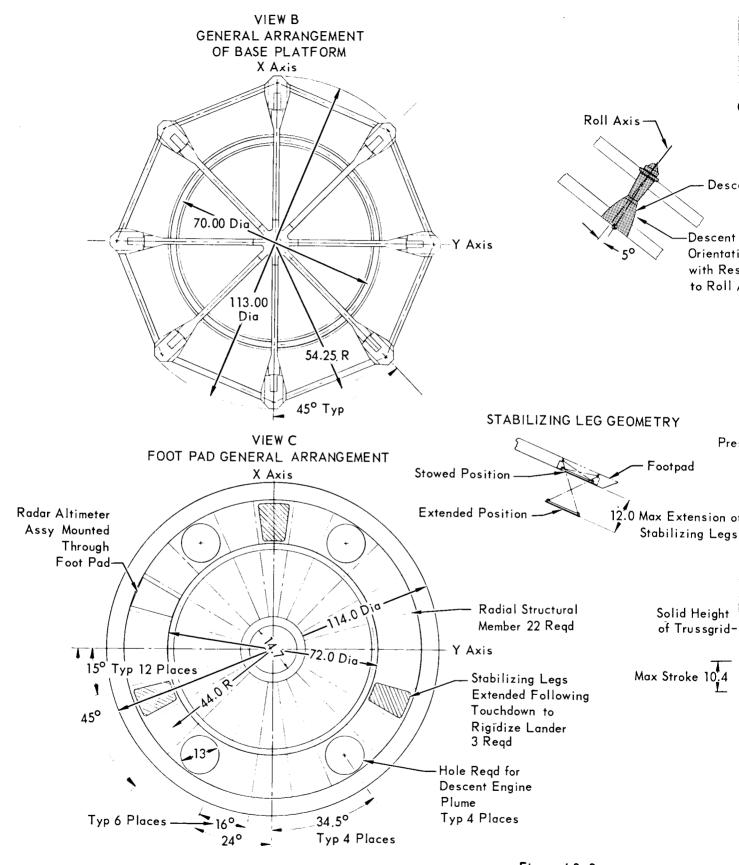
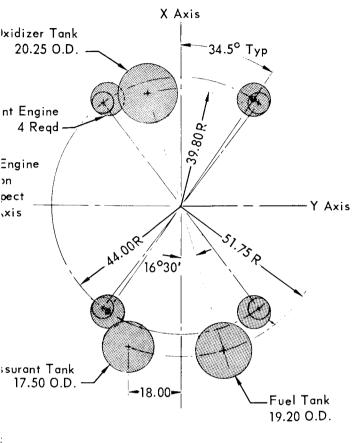


Figure 4.2-2 (Continued)

4-20 -

VIEW D
PROPULSION SUBSYSTEM GENERAL ARRANGEMENT



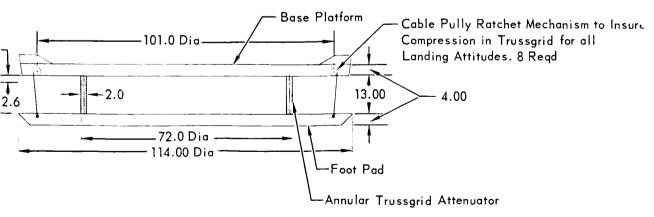
VIEWS B & D

VIEW C

VIEW C

LANDER CAPSULE

VIEW E LANDER IMPACT MECHANISM GENERAL ARRANGEMENT



VIEW F
TELEMETRY SUBSYSTEM GENERAL ARRANGEMENT

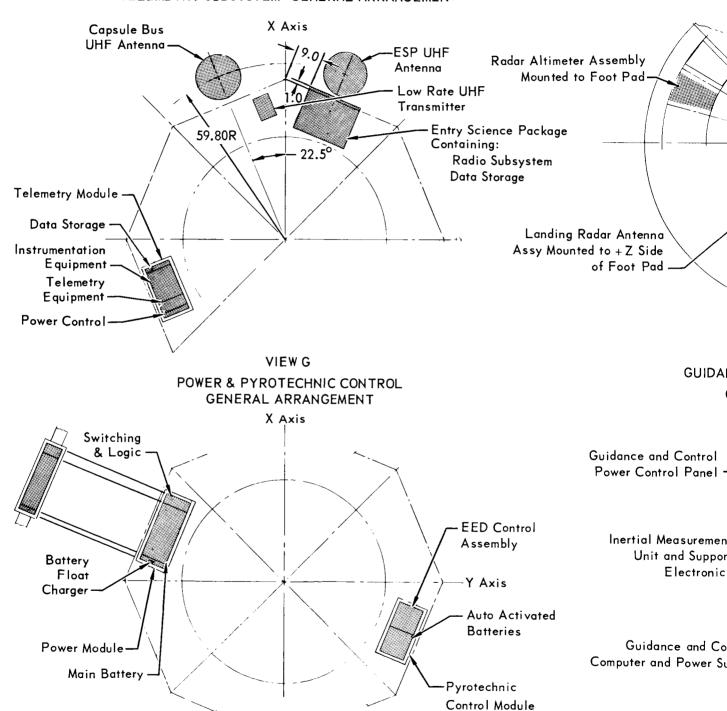


Figure 4.2-2 (Continued)

4-21-1

AR GENERAL ARRANGEMENT X Axis Power Control Unit Sequencer & Timer Radar Electronic Assembly Y Axis Radar Sequencer & Timer Module VIEWS F, G,H & IT VIEW I ICE AND CONTROL SUBSYSTEM ENERAL ARRANGEMENT X Axis CAPSULE LANDER Y Axis htrol pply

VIEW H

aluminum Trussgrid crushable cylinder sandwiched between the large flat circular footpad and the base platform structure. This is a completely passive system, being stowed in the landing condition. There are eight tension cable pulley assemblies connecting the base platform and the impact footpad. Their function is to insure that the aluminum Trussgrid attenuator feels only compression and shear forces, no tension.

4.2.1.1 Physical Description - The landing footpad is 114 inches in diameter and 4 inches thick, with a turned-up outer lip to facilitate sliding over small obstacles. The footpad is made up of titanium radial beams, rings, and lower skin. Sufficient structural rigidity is required in this pad to distribute impact loads uniformily to the crushable attenuator when landing on one edge or on peaks or ridges. The terminal descent engine nozzles thrust downward through four 13 in. diameter holes in the footpad. These holes are between the radial beams of the pad at a radius of 44 inches.

The base platform is made up of eight titanium I-beams, 4 inches deep, with 2.5 inch caps, equally spaced in a radial spoke arrangement. The maximum platform diameter is 113 inches.

The impact attenuator is installed between the landing footpad and the base platform. It is a crushable aluminum (Trussgrid by American Cyanamid Co. or Cross-Core by Hexcel Products) cylinder, 13 inches high, 72 inches outside diameter, and 2.1 inches wall thickness. This Trussgrid is bonded and mechanically keyed to both the footpad and the base platform through structural channel rings to insure the transfer of landing loads. The Trussgrid used is:

.003 thick foil 5052 - H339 aluminum 3.3 lb/ft³ density 75 psi crush strength

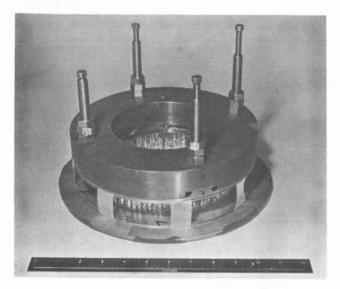
Eight tension cable pulley assemblies are mounted one each to the ends of the four radial I-beams of the base platform outboard of the Trussgrid attenuator. These cables tie the footpad to the base platform and serve as pivot points for the rigid footpad when it lands on the opposite edge. By forcing the footpad to rotate about this point, the entire crushable attenuator is put into compression, eliminating tension loads across the bond between attenuator and adjoining structure. As the Trussgrid crushes, the spring loaded cable pulley assemblies take up the cable slack; one-way ratchets prevent cable lengthening. This insures a repeat attenuation capability should the lander bounce and land on the opposite edge.

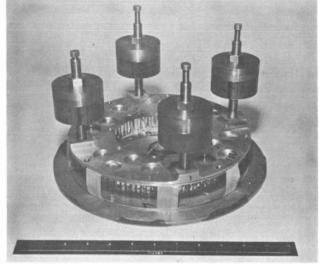
In addition, this continuous snugged-up condition of the structure and the Trussgrid insures attenuation of the horizontal loads through the Trussgrid. (These horizontal loads may also be dissipated in part through friction between the footpad and the Mars surface when sliding occurs.)

4.2.1.2 - Advantages/Disadvantages - The Uni-Disc Lander is the only configuration of the many that were studied that meets all the constraints set forth in the Phase B study constraints document, as well as our self-imposed requirements. With this configuration, it is necessary to crush terminal descent engine nozzles and landing radar antenna on impact. This disadvantage is countered by a substantial list of advantages:

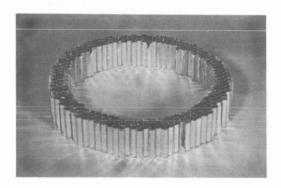
- Meets or exceeds all landing constraints.
- Large footprint area for soft soil conditions.
- Best accessibility to equipment and Surface Laboratory.
- Simple passive landing system which is stowed in the landing condition,
 therefore, not requiring deployment after Aeroshell separation.
- Highly flexible Surface Laboratory installation plus growth capability to full mobility.
- Good c.g. location in Aeroshell.
- Moderate landing gear weight.
- Short, easy reaches for sample gatherers, etc.
- Ample unrestricted radiator surfaces.
- Simple, straight forward structural design using all state-of-the-art materials and manufacturing methods.
- Low development cost and risk.
- Shape conducive to clean separation from the Aeroshell.
- Landing system should meet no adverse problems during sterilization or exposure to hard vacuum for long periods.
- Terminal descent engine installation which does not impinge on any other portion of the lander.
- Good landing radar installation (bottom centerline).
- 4.2.1.3 <u>Test Program of Uni-Disc</u> A 1/10 scale drop test model (Figure 4.2-3) of the Uni-Disc Lander was built, with an overall weight of 27 lbs., of which 24 lbs. was the attenuated portion and 3 lbs. was the unattenuated weight of the footpad and attach hardware. This ratio closely simulates the ratio of the full-scale lander. Sandpaper was bonded to the underside of the footpad, and a 1/8 inch thick sheet of hard rubber was bonded to the 34° slope. This combination produced a static coefficient of friction of 1.0. Dynamically this friction coefficient on impact was

UNI-DISC LANDER DROP TEST HARDWARE MODEL SCALE 1/10

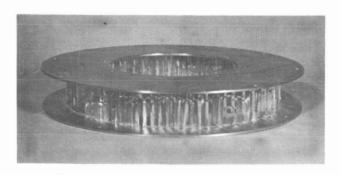




UNI-DISC MODEL DROP CONFIGURATIONS



ALUMINUM HONEYCOMB ATTENUATOR



PRE-CRUSHED ATTENUATOR INSERT



ATTENUATOR INSERT WITH TENSION STRAPS (POST DROP NO. 19)

Figure 4.2-3

4 - 24

more nearly 1.5 to 2.0.

The impact attenuator used was a hollow cylinder of 5052 aluminum hexcel honeycomb having 6 inch inside diameter and a 1.0 thick wall or a .8 inch thickness. The crushable cylinder was 1.30 in. high and bonded between two 0.08 in. thick sheets of aluminum which were bolted to the footpad and lander base platform. The honeycomb had 1/4 in. cells of perforated foil .0009 in. thick. The crushing strength was 55 psi. Figure 4.2-4 is a representative list of drop tests and results and Figure 4.2-5 shows sequential frames from high speed movies of a few of the tests.

Drop #42 was a test of stability in a down-hill slide. For this test, the friction surface was changed from sandpaper/rubber to an aluminum incline/teflon-coated footpad. The sliding coefficient of friction was 0.22. The model was hand-released from various distances up the 34° slope and allowed to slide into a fixed obstruction. With the model c.g. height at 2.6 in., the slide distance was increased until turnover occurred. At a slide distance of 25 1/4 in. (a simulated impact velocity of 17.4 fps), the Uni-Disc was stable. At a slide distance of 28 1/4 in. (simulated impact velocity of 18.1 fps) the model was unstable. Successful repetition of these runs established high confidence in these results.

The results of this drop test program proved decisively that the Uni-Disc Lander exceeds the landing stability requirements of the Phase B constraints document.

The predicted c.g. heights above the bottom of the footpad are 24.1 inches for the full scale 1973 lander and 27.9 inches for a 1979 Uni-Disc with a six wheel, fully mobile Surface Laboratory aboard. The 1/10 scale model tested was stable up to and equivalent c.g. height of 32.4 inches.

A computer program was developed to simulate the Uni-Disc landing characteristics and the results of many runs confirm all of the test results. These programs establish a very high confidence level in the Uni-Disc lander configuration.

4.2.2 Phase B Study - The purpose of this phase of the study was to evaluate the candidate configurations surviving the early (pre-Phase B) study evaluation, but

4.2.2.1 <u>Technical Requirements</u> - The major functions of the systems which comprise the Capsule Lander are:

Entry Science Package

under the new constraints.

o Perform all entry science experiments and transmit acquired data.

1/10 SCALE UNI-DISC LANDER MODEL DROP TEST RESULTS

REMARKS	Stable	Stable	Stable	Stable	Stable	Stable upper limit of C.G. for sta- bility	Unstable	Stable – 10° nose high	Stable	Stable; impact with protrusion while sliding down 34° slope impact velocity = 17.3 fps relative to slope	Stable – 1st impact downhill
ATTENUATOR WALL THICKNESS (IN.) 6" ID. 1.3" HEIGHT	1.0	1.0	1.0	1.0	1.0	1.0	1.0	0.8	0.8	8.0	0.8
HORIZONTAL VELOCITY – SIMULATED (FPS)	23	10	10	10	10	10	01	01	0	I	10
VERTICAL VELOCITY - SIMULATED (FPS)	91	16	91	25	16	91	16	16	25	I	16
C.G. HEIGHT FROM BOTTOM OF FOOTPAD (IN.)	2.6	2.6	2.6	2.6	2.76	3.24	3.35	2.6	2.79	2.6	2.6
TOTAL MODEL WEIGHT (LB)	26.875	26.875	26.875	26.875	26.875	26.875	26.875	26.875	26.875	26.875	26.875
VALLEY											×
RIDGE		×									
OPE UP- HILL IMPACT			×								
34° SLOPE DOWN- HILL HILL HMPACT IMP	×			×	×	×	×	×	×	×	
DROP TEST NUMBER	12	91 _*	×17	61*	23	30	29	*37	38	* 42	*45

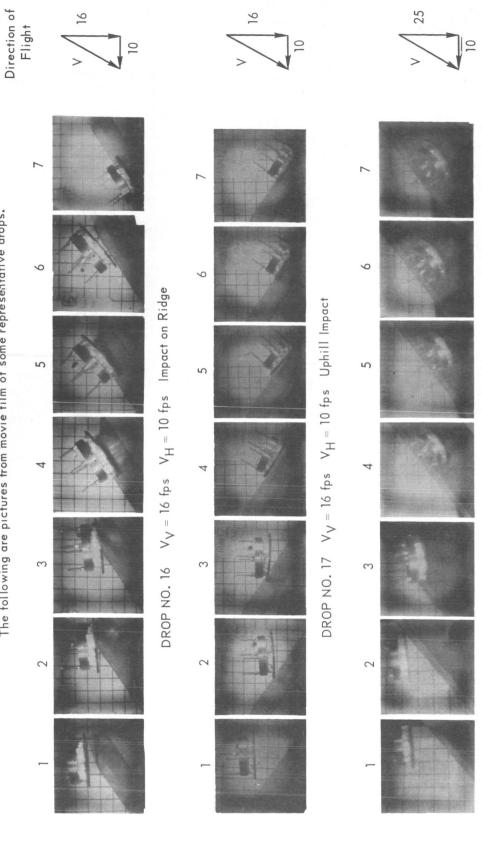
*Sequential photographs of these drops appear in Figure 4.2.5

** A For Detailed Explanation See Text

Figure 4.2-4

UNI-DISC DROP TESTING 1/10 SCALE MODEL





DROP NO. 19 $\text{ V}_{V} = 25 \text{ fps } \text{ V}_{H} = 10 \text{ fps } \text{ Downhill Impact}$

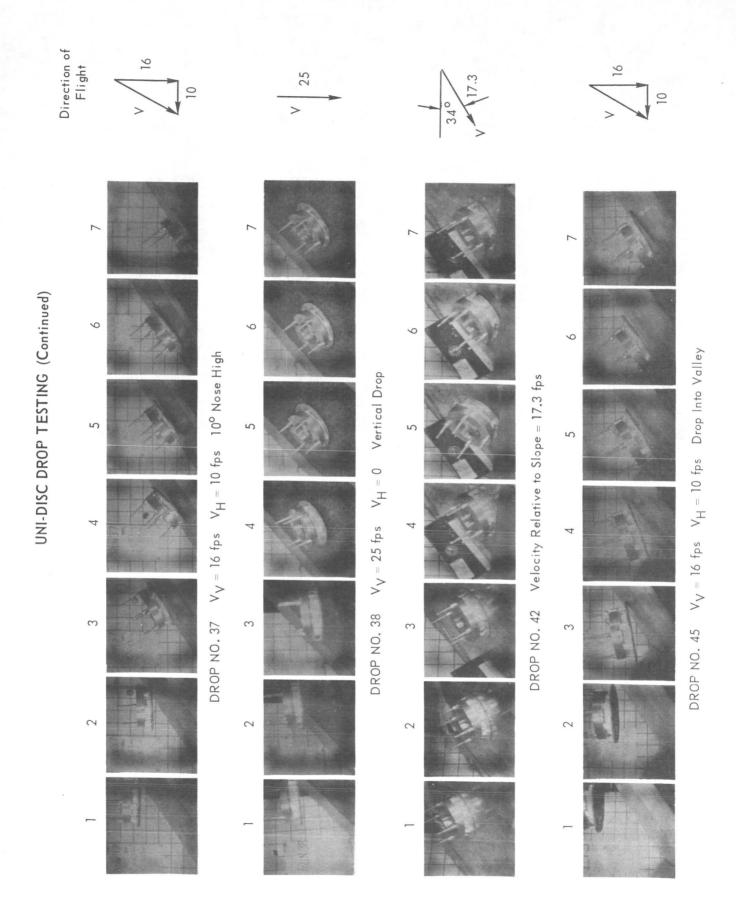


Figure 4.2-5 (Continued)

Surface Laboratory

- Measurement and transmission of engineering performance data.
- Update control sequences upon command.
- Perform landed science experiments and transmit acquired data.

Capsule Bus

- Measurement and transmission of engineering data from spacecraft/capsule separation through the landing phase.
- Delivery of the Entry Science Package into the Mars atmosphere.
- Landing and erection/stabilizing of the Surface Laboratory on the Mars surface.
- Turn self off after landing and after transmission of data to ascertain that its function was performed.

4.2.2.1.1 <u>Design Constraints</u>

- To the extent practical, the Capsule Bus, Surface Laboratory, and Entry Science Package shall be mutually independent, separable and self-supporting.
- Capsule Bus/Surface Laboratory and Capsule Bus/Entry Science Package physical interfaces shall each consist of a structural field joint and an electrical connection.
- The Capsule Bus shall be standardized and compatible with the requirements of 1975, 1977, and 1979 Mars opportunity and shall be designed to maximize allowable Surface Laboratory plus Entry Science Package weights.
- Emphasis on simple and conservative design.
- Capsule Bus shall be a fully automated device designed to de-orbit, enter, descend, and land.
- Surface Laboratory structural load during landing shall not exceed the load induced during earlier phases of the mission.

4.2.2.1.2 Weight Constraints

- Total Flight Capsule weight is 5000 lbs. for 1973, 7000 lbs. for later missions.
- Provide weight contingency.
- The combined Surface Laboratory and Entry Science Package weight shall not be less than 900 lbs.

The preliminary design weights are as follows:

	1973-1975	1977-1979
Flight Capsule Weight	5000	7000
Pre-de-orbit Weight	4300	6170
Post-de-orbit Weight	3835	5500
Entry Weight	3680	5390
Ballute Deployment Weight	3680	5390
Terminal Propulsion Initiation Weight	2650	4165
Touchdown Weight	2260	3560
Surface Laboratory Weight	1400	1890

4.2.2.1.3 <u>Volume Requirements</u> - A packing density of 35 lb/cu ft., derived from previous McDonnell designs such as Gemini, Asset, Mercury, F-4, and Model 122, was used for the 1973 Surface Laboratory.

The minimum Surface Laboratory weight (including core group experiments both descent and landed) is 900 lbs. The weight and volume was distributed as follows and as graphically represented in Figure 4.2-6. (Note: these values have changed since this study was conducted. The study conclusions are not affected.)

Density		
<u>1b/ft³</u>	<pre>% Volume</pre>	% Weight
54.6	5.6	9.2
86.4	0.33	0.88
34.6	12.0	12.5
88.2	9.4	24.9
207.0	2.8	17.7
6.9	63.0	12.9
172.8	2.4	12.5
-	4.3	9.2
	1b/ft ³ 54.6 86.4 34.6 88.2 207.0 6.9	1b/ft3 % Volume 54.6 5.6 86.4 0.33 34.6 12.0 88.2 9.4 207.0 2.8 6.9 63.0 172.8 2.4

SUBSYSTEM WEIGHT AND VOLUME ALLOCATIONS (900# LAB)

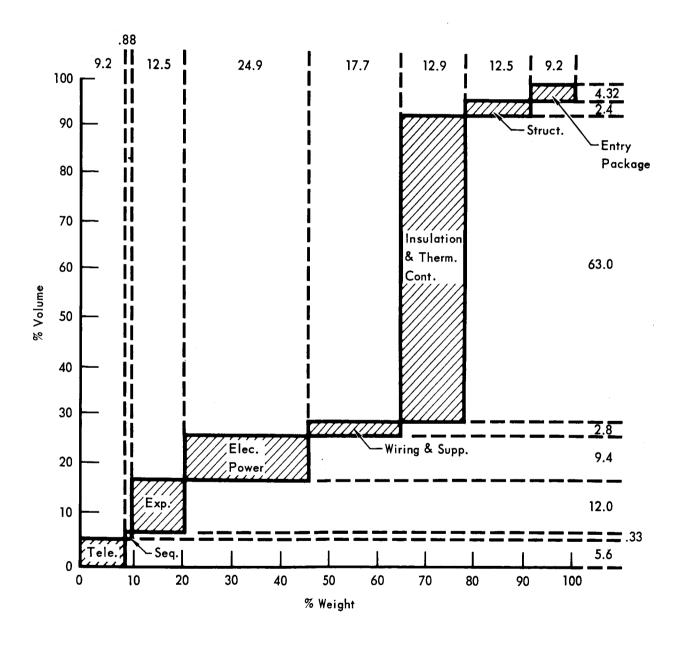


Figure 4.2-6

4.2.2.1.4 <u>Study Guidelines</u> - The following guidelines for each system were used: Surface Laboratory

•	General	<u>1973</u>	<u>1979</u>
	Weight, 1b.	900	1890
	Density, 1b/ft ³	35	35
	Volume, ft ³	25	34
	Radiator Surface	19 ft^2	
	Heat Dissipation	230 Watts (approx.)	
	Mobility	Stationary with	Fully mobile
		possible mobile	
		sample gatherer	
		(dependent 4 wheel	drive)

- Leveling Requirements None (individual subsystem component leveling if required).
- 20g maximum load during landing
- Functional scientific and data compartment if mobility failure occurs.
- RTG electrical power shall be used on the 1977-79 mobile Surface Laboratory.
- 1979 Mobile Surface Laboratory Weight Breakdown:

Weight	1890
Mobility (6 wheels @ 40 each)	-240
Structure	-216
Electrical Power (300 Watt RTG)	-300
Scientific & Data Compartment	1134
1973 Surface Lab Electrical Power	<u>-700</u>
Growth Capability	434 1b

Capsule Bus

- Omnidirectional within $\pm 45^{\circ}$ of vertical
- Surface Laboratory mobility: 1973-partial
- Surface Laboratory mobility: 1979-full

- Surface conditions:
 - 34° Maximum slope 100 meters long 68° Abrupt surface discontinuity (includes conical) (See Figure 4.2-7 for surface landing geometry)
- 6 lb/in² to infinity soil bearing capacity
- 12.5 cm (5") rocks
- 20g maximum load to laboratory during landing
- Descent propulsion 3 engines if possible, 4 otherwise

Landing Gear Design Conditions - Figures 4.2-8 and 4.2-9 show the landing conditions imposed on the Stabilized Platform with outriggers both low and high. Figure 4.2-10 shows the landing design conditions for the four legged composite lander. (The Stabilized Platform and four legged composite lander are described in Section 4.2.2.2).

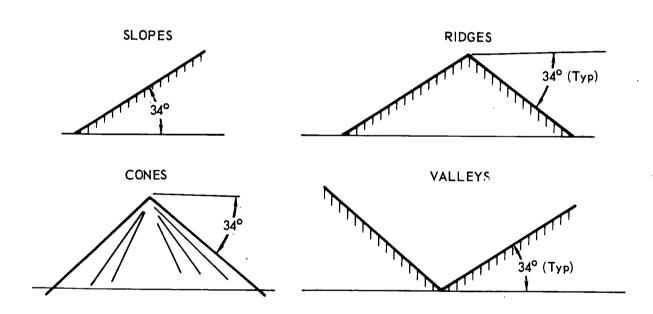
Antenna Installation

• Landing Radar	24" x 26" x 6.5" 13" x 8" x 6.5"	Lower Surface C Adjacent
Ranging RadarAltimeter ArrayVHF (2)	36" Dia. Parabolic 4" x 8" x 3/4" 15" Dia. x 5"	£ of Aeroshell Aeroshell 21" Dia. Ground Plane
		Operate During Descent and Landing Phases
• S-Band High Gain	18" x 18" x 2.5"	Erectable with Tracking Capability
• S-Band Omni	5-1/4" Dia. x 2.3"	Top Surface

<u>Deployed Experiments</u> - Figure 4.2-11 shows view angles of cameras and areas of surface contact of sample probes.

- 4.2.2.2 Design Approaches The Phase B study considered three configurations:
- (1) the Torus Rover Lander, which scored highest in the early study evaluation,
- (2) The Composite Legged Lander, which combined the best features of some of the configurations scoring well in the early study evaluation, and (3) the Stabilized Platform, which was designed specifically in response to the Phase B constraints (Reference 4.2-2). As the study progressed, the Stabilized Platform Lander evolved into the preferred Uni-Disc design.
- 4.2.2.2.1 <u>Torus Rover</u> The Torus Rover Lander (Figure 4.2-12) studied in the pre-phase B study (see 4.2.3.3.6) was re-evaluated with respect to the new constraints. The following problem areas were identified:

SURFACE LANDING GEOMETRY CONSTRAINTS



		COMMENTS	All Points Contact Simultaneously, Critical for Load Factor	Outriggers Contact First.	Load and Stroke on	One Leg				Crifical for Stability
INDITIONS	ERS)	SKETCH	THE STATE OF THE S			KKKKK THE STATE OF	-	Miller	-	KLALL
LANDING DESIGN CONDITIONS	(LOW OUTRIGGERS)	FRICTION	N/A	N/A	-	High	i	ngiri		High
LANDIN	1)	SURFACE	Conical Depression	Conical Depression	, c	34° Scope	000	34 Cone	C	34 Scope
		HORIZONTAL VELOCITY	0	0	5 FPS	10 FPS	5 FPS	10 FPS	5 FPS	10 FPS
		VERTICAL VELOCITY	20 FPS	20 FPS	20 FPS	16 FPS	20 FPS	16 FPS	20 FPS	16 FPS
		CONDITION	-	2		m		4	i.	n

Figure 4.2-8

LANDING DESIGN CONDITION (HIGH OUTRIGGERS)

COMMENTS	Critical for Maximum Load Factor		Critical for Stability		Critical for Side Loads	
SKETCH						
FRICTION	N/A		dg:	High		
SURFACE	Level	Cone	Nith -34 Slopes	0	SA Scope	
VERTICAL HORIZONTAL	0	5 FPS	10 FPS	5 FPS	10 FPS	
VERTICAL VELOCITY	20 FPS	20 FPS	16 FPS	20 FPS	16 FPS	
CONDITION	_	C	7	c	2	

Figure 4.2-9

LANDING GEAR DESIGN CONDITIONS

(4 LEGGED LANDER)

COMMENTS		Max. Lodd racior	Critical for Strength and Energy in One Leg	Observation of Results May Eliminate the Need for Some of These			.1. 1 3 7 1 1 22 2	Critical for Stability	
SKETCH	-							-[]	
CONTACT	2		1-2-1	1-2-1	2-2	1-2-1	2-2	1-2-1	2-2
FRICTION	High	0	High	Ţ	- -	c	>	[20
SLOPE DEGREES	C	•	34		37			76	5
HORIZONTAL VELOCITY FPS	L.	•	10	٠c		01			
VERTICAL COND. VELOCITY FPS	00	2	16	20			91		
COND.	-	-	2		m)		4	

Figure 4.2-10

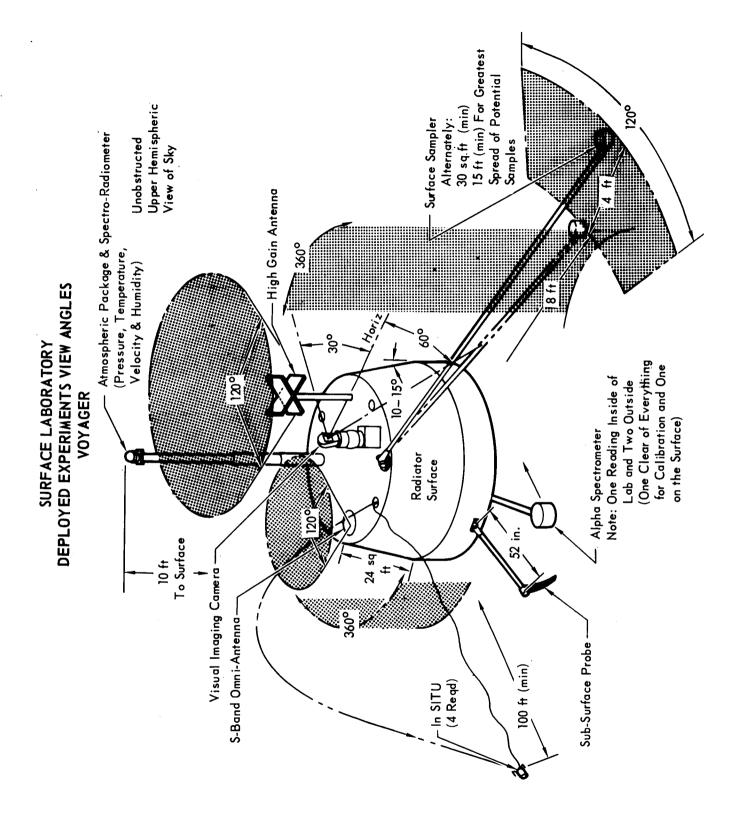
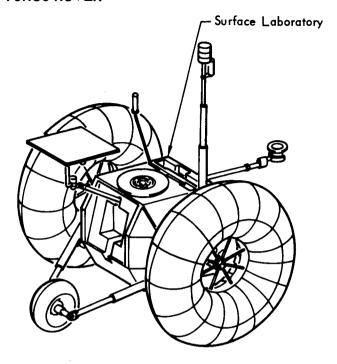


Figure 4.2-11

GENERAL ARRANGEMENT DUAL TORUS ROVER



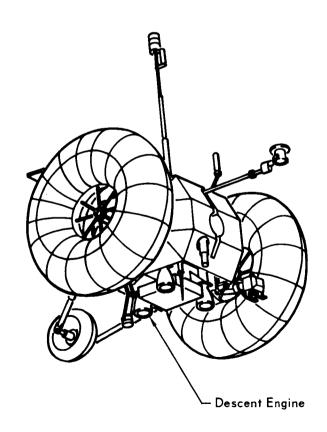


Figure 4.2-12

4-39

Geometry

- 1. Limited radiation surface area
- 2. De-orbit motor loads taken through Surface Laboratory structure
- 3. Limited view angles for visual and antenna systems
- 4. Employment of an extendable torque arm as a function of both packaging and mobility.
- 5. Requirement for an auxiliary impact attenuator to accommodate peaked terrain landing conditions.
- 6. Descent equipment mounting area limitations imposed by the auxiliary impact attenuator.
- 7. Payload volume limitations from 360° ridge clearance requirements during tumbling together with upper limits of tori sized for practical installation.
- 8. Requirements for fully deployable and retractable experiments because of tumbling or payload rotation.
- 9. RTG installation, separation, and clearances.
- 10. Growth capability restrictions due to volume limitations.

Alternatives - 7, 8, 9, 10. Minimize bounce by venting the tori and provide stabilization arms to prevent lander tumbling; provide both fore and aft torque arms to limit payload rotation during roving.

• Orbit and Descent

- 1. Effects of long term storage in folded configuration.
- 2. Meteorite penetration after separation from Spacecraft.
- 3. Entry heat effects on tori pressures.
- 4. Rapid inflation techniques and temperature effects.
- 5. Narrow tolerance band for required inflation pressure.
- 6. Descent engine exhaust impingement.

Alternatives - 3, 4, and 5 employ torus pressure regulation subsystem.

Impact

- 1. Possibility of puncture from sharp pertuberances.
- 2. Roving capability jeopardized by damage incurred during landing A single torus failure which would terminate roving capability points out the need for a back-up secondary bladder.
- 3. Configuration is susceptible to bouncing which would increase the likelihood of tumbling.

- 4. Increased secondary impact loads are possible should downslope tumbling occur.
- 5. Torque arm is vunerable to damage during tumbling.
- 6. Limiters required to prevent coming to rest on arm.
- 7. Vulnerability of loading torque arm to obstacle damage during roll-out braking.
- 8. Possibility for an adaptive braking subsystem to meet both up and down slope landing conditions.
- 9. Possibility of rolling or sliding to a low terrain feature limiting field-of-view and communications.
- 10. No protection from impacting on cone.

Alternatives

3. Utilize compartmentation and internal venting to minimize rebound and subsequent tumbling.

Landed Operations

- 1. Possibility of puncture from sharp pertuberances or abrasions.
- 2. Impact pressures are much higher than those required for optimum tractive capability.
- 3. Solar radiation-induced pressure changes may present minor stable platform problems.
- 4. Large areas and gas suspension may make Laboratory susceptible to wind load vibrations and require the use of stabilization struts.
- 5. Possible problem of torus material embrittlement from low temperature extremes.
- 6. Baseline concept has step climbing capability of 1/10 torus dia. and obstacle capability equal to the trailing wheel radius.
- 7. Baseline concept has a maximum slope climbing capability of 24 1/2 degrees on a hard surface and less for soft surfaces.

Alternatives

6 and 7. Provide power to the trailing wheel; increase wheel base to approximate the tori tread dimension; and finally, increase trailing wheel diameter to 51 in. for one meter obstacle override capability or to 70-75 in. for optimum tori frictional coefficient (G.M. data based on 144 in. tori diameter).

• Development Items

1. Materials

- a. Composition laminates; composites; welding; bonding
- b. Chatacteristics abrasion; puncture; tear; flexure; permeability; storage and operating life.
- c. Environment temperature; vacuum; chemical; radiation
- 2. Pressurization subsystem
- 3. Pressure regulation and replenishment subsystem
- 4. Impact Dynamics
 - Impact attitude; auxiliary impact attenuation; pressures; c.g. location
 - b. Bounce stabilization; loads, internal venting; dual pressure systems
 - c. Roll-out stabilization; braking mode

5. Performance

- a. Obstacle clearance and override capability
- b. Slope climbing capability
- c. Static and dynamic instability

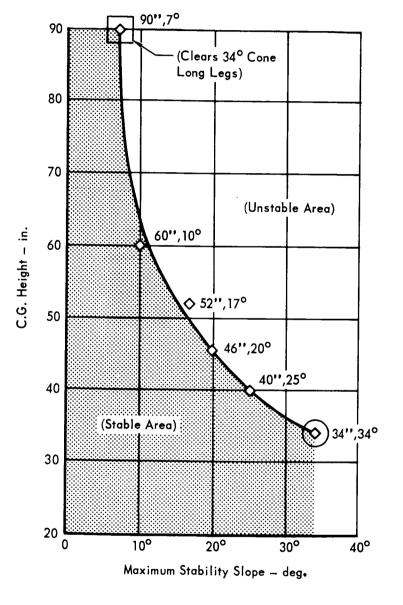
4.2.2.2.2 <u>Modified Composite Legged Lander</u> - At the end of the pre-Phase B study, it was decided to continue study on a composite legged lander configuration which adopted some of the best features of the legged configurations studied. Our computerized stability calculations indicated that four legs were needed to maintain stability on a 34° slope if the span of the gear and the height of the c.g. were unchanged from the three legged version originally configured (Section 4.2.3.3). This configuration was capable of landing on ridges with 32° abrupt changes in slopes and cones with slopes of 11°, as indicated in Figure 4.2-13.

The 1973-75 version of this lander, shown in Figure 4.2-14 has the Surface Laboratory broken up into four basic packages mounted to the top outboard edges of the Capsule Bus platform with provision in the center for a small dependent rover laboratory.

The 1977-79 version of this lander as shown in Figure 4.2-15 has a completely mobile rover mounted on top of the capsule platform. This Surface Laboratory is capable of functioning as-landed should the roving deployment fail.

Figure 4.2-13 shows the stability envelope of the four legged geometry studied together with the condition used.

FOUR LEGGED LANDER (STABILITY ENVELOPE)



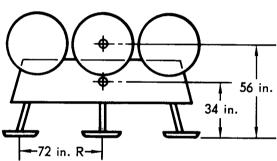
Fixed Condition

= 465 Slugs ft/sq

V_V = 16 ft/sec V_H = 10 ft/sec W_T = 2260 lb

Time After Initial Impact = 3 sec

Down Slope Impact



Capabilities

1. Abrupt Change in Slope = 32°



2. Maximum Cone = 11°



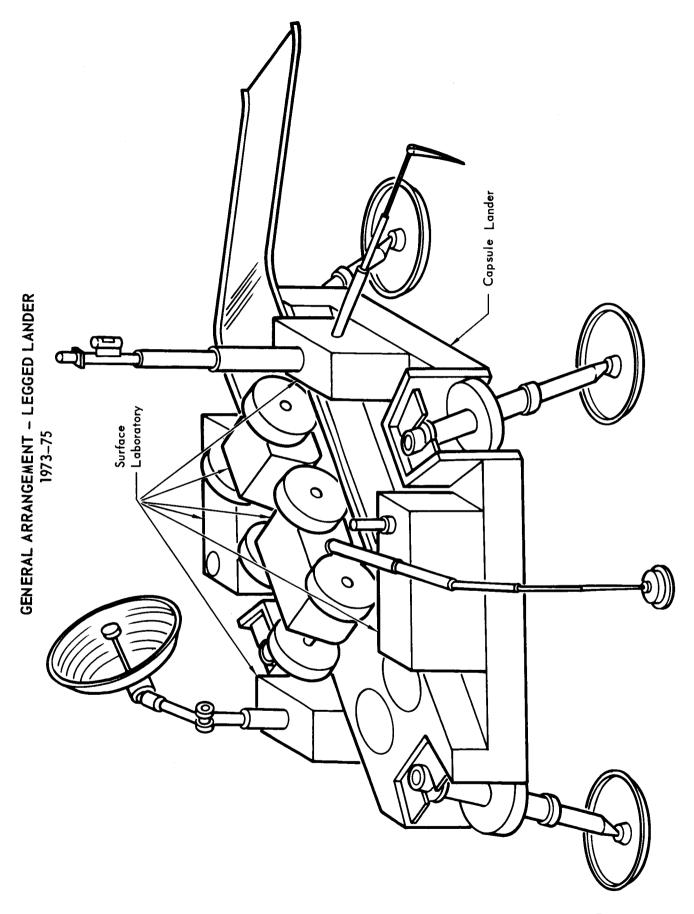
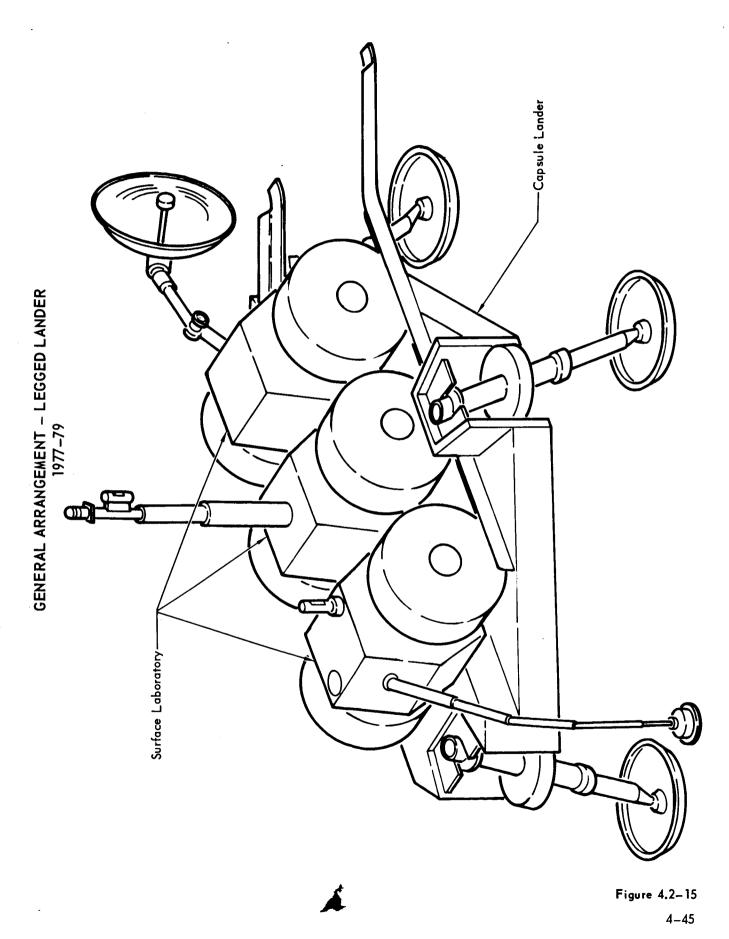


Figure 4.2-14

4-44



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4.2.2.1.3 <u>Stabilized Platform</u> - The Stabilized Platform configuration was developed specifically to meet the new constraints received with the Phase B contract. It incorporates many of the features of the composite legged lander. Four different four arm concepts were considered (shown in Figure 4.2-16).

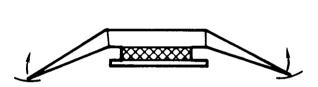
The main impact energy is taken in the central pad, outriggers being used only to increase stability in roll-over conditions and thus absorbing very little energy. With arms low, during a landing made in a valley, the arms must take all the landing load. If they are sized to do this, when a landing is made on a slope, the arms impart large overturning moments. With arms high, valley landings can be made but the arms are too high to be of use in roll-over on the cone. The third concept mounted the arms high but as soon as the main pad felt load the legs would be released and driven down. If the arms could be actuated with enough force to move them quickly enough into position to be of assistance in overturning, they would feed large dynamic loads into the lander, decreasing stability. The fourth concept was an attempt at correcting the faults of the first three. In order to reduce the overturning moment of the uphill arm, each arm has the ability to stroke at two different rates.

All arms are initially down and resist movement with a small force. When an arm strokes, it rigidizes the arm opposite which then requires larger forces to stroke. Thus, the large overturning moment of the uphill arm is eliminated and fast motions of large arm assemblies are not required. The problem with this system is that the lander may rotate during landing, placing the arms in their worst attitude for stability.

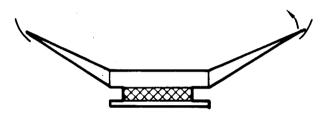
All four arm systems suffer a large weight penalty because the arms must be quite long to be of value during roll-over conditions on slopes and cones. The general arrangements of 1973 Stabilized Platform Landers are shown in Figure 4.2-17. The arrangement of the 1973-1975 lander is shown in Figure 4.2-18. The arrangement of the 1977-1979 lander is shown in Figure 4.2-19.

4.2.2.3 Evaluation and Recommendation - The Torus Rover Lander, which scored highest under the old constraints, was eliminated from consideration by the new surface constraints. The Composite Legged Lander underwent considerable change after receipt of the new constraints (4 legs and new Surface Laboratories) but was still unable to land on cones and ridges; it was also eliminated. Many of the good features of the Composite Lander appear in the Stabilized Platform Lander. This concept could probably have been developed to meet the constraints but the complication of the mechanized outriggers and the associated weight seemed excessive. This configuration has a very low center of gravity, and can accommodate a wide variety of Surface Laboratory Shapes. As work progressed on the platform lander, computer

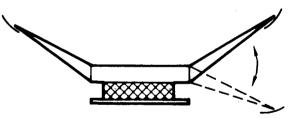
OUTRIGGER ARM CONFIGURATIONS STUDIED FOR STABILIZED PLATFORM LANDER



1. Arms low, soft resistance varying to stiff



2. Arms high, only used when in extreme angles

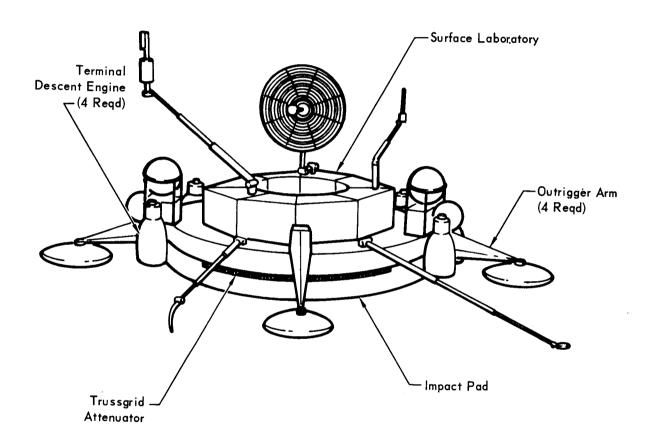


3. Arms high until center pad contact



Figure 4.2-16

GENERAL ARRANGEMENT 1973 STABILIZED PLATFORM LANDER



GENERAL ARRANGEMENT 1973-75 STABILIZED PLATFORM LANDER WITH DEPENDENT ROVER

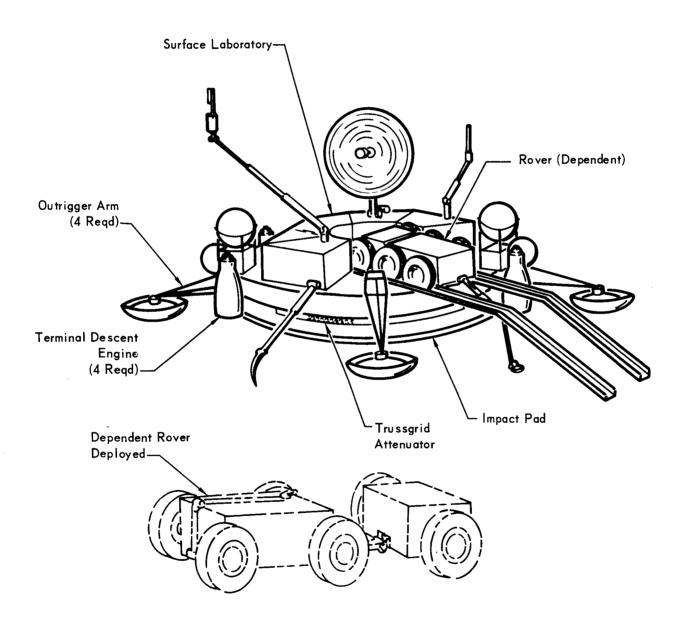


Figure 4.2-18

4-49

Figure 4.2-19

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analysis showed that if the landing pad was large and the center of gravity low, outrigger arms were not necessary for landing stability on 34° slopes, cones, ridges, and valleys. This, then, became the Uni-Disc Lander, and was selected as our preferred design.

- 4.2.3 Pre-Phase B Study This portion of the study is included for historical completeness and to show how the earlier study helped to configure the newer design candidates the Composite Legged Lander and the Stabilized Platform. The purpose of this study was to generate feasible Capsule Lander configurations, which entails installation into the Aeroshell, Surface Laboratory compatibility for 1973 thru 1979, and landing constraints effect. Six configurations were evaluated from which three of the most feasible configurations were chosen. The evaluation encompassed: (1) omni-directional, (2) mono-directional with tumbling capability, and (3) mono-directional Capsule Landers.
- 4.2.3.1 <u>Technical Requirements</u> Except for the volume requirements and the study guidelines, the requirements were the same as those reported in 4.2.2.1.
- 4.2.3.1.1 <u>Volume Requirements</u> The 1973 Surface Laboratory preliminary packing density was 35 lb/cu. ft. The first cut of the Surface Laboratory equipment weight was approximately 700 lbs., including wiring and structure. The volume required for the 1973 Surface Laboratory is then 20 cu. ft. The densities of the Surface Laboratory subsystems are:

	Density, 1b/ft ³	% of Total Volume
Telecommunications	54.5	10.4
Electrical (Batteries)	91.7	8.1
Experiments	31.3	22.5
Insulation (3")	15	47.7
Structure	60.0	10.8

4.2.3.1.2 <u>Study Guidelines</u> - The following guidelines for each system were adhered to:

Surface Laboratory

•	<u>General</u>	700 lb.	1400 1b.
	Density, 1b/ft ³	35	35
	Volume, ft ³	20	40
	Radiator surface, ft ²	17*	24
	High "9" equipment volume, ft ³	9	13
	Low "9" equipment volume, ft ³	11	27
	External equipment volume, ft ³	3.3	4.5

^{*} Approximately 200 Watt Heat Dissipation

- Exterior radiator surfaces should be canted 10-15° to vertical and should have a clear field of view.
- The high heat output equipment compartment shall be located above the low heat output equipment compartment.
- Total Surface Laboratory shall be leveled to within 3° of the local vertical. Capsule Bus
 - Stow in 19', 60° aeroshell
 - $v_{\text{vert}} = 20 \text{ fps } e^{V_{\text{hor}}} = 0$, $v_{\text{vert}} = 10 \text{ fps } e^{V_{\text{hor}}} = 16 \text{ fps}$
 - Surface Conditions 30° maximum slopes, 10 cm rocks
 30° abrupt surface discontinuity
 - 3.0 lb/in²/ft width (used to size foot pads)
 0.43 to 1.05 lb/in²/ft bearing capacity (per Constraints
 Document)
 - 20g maximum on payload during landing.
 - Accommodate 700 1b and 1400 1b Surface Laboratory with standard interface.
 - Entry Science Package shall be attached to Capsule Bus.

 Landing gear design conditions are shown in Figure 4.2-20.
- 4.2.3.2 <u>Design Approaches</u> A total of fifteen candidate lander configurations were considered. Nine were rejected after a preliminary investigation, six were studied in depth.
- 4.2.3.2.1 <u>Legged Configuration</u> The Capsule Lander is a three-legged, mono-directional lander capable of landing a 1270 lb Surface Laboratory on Mars. The three legs are stowed with the main struts folded. The legs are unfolded during terminal descent and locked in the down position by means of a mechanical lock. The side braces are used to attenuate horizontal loads through double acting cylinders containing a crushable balsa core. Vertical landing shocks are absorbed by crushable balsa cores in the main struts. A tubular truss structure is used to support the legs and braces. This truss is attached to the lander primary structure.

The Capsule Lander primary structure consists of three conventional cap and web beams 6 in. deep and 100 in. long. The three beams form a triangle with the apexes being the three attach points to the Aeroshell. The upper caps of the beams are connected by a flat shear web. Local stiffeners are provided for mounting the terminal propulsion fuel, oxidizer, and pressurization tanks with their associated

LANDING GEAR (3 LEGS) DESIGN CONDITIONS

COMMENTS	Max. Load Factor	Critical For Clearance	Critical For Energy in One Leg.	Observation of Results Will Eliminate The Need	Critical For Stability	
SKETCH	+	mmm		A E		
CONTACT SEQUENCE	e co		1–2	1-2	1-2	.1-2
FRICTION	High	0	High	High	0	High
SLOPE DEGREES	c	-	20	ç	30	
HORIZONTAL VELOCITY FPS	c	>	01	0 01-		-10
VERTICAL VELOCITY FPS	ç	2	91	ć	91	
COND.	-	_	8		ю	

Figure 4.2-20

hardware; the Capsule Bus lander equipment; the Entry Science Package and telemetry equipment; and a platform for the Surface Laboratory. The fuel and oxidizer tanks are mounted inboard of the beams below the shear web. The pressurization tanks are mounted on the top of the shear web outboard of the Surface Laboratory. The other lander equipment is mounted above and below the shear web.

Terminal propulsion is provided by six liquid fueled descent engines. The descent engines are mounted in three pairs, midway between the Aeroshell attach points, on the outboard side of the beams, between the landing leg braces.

The 58 in. diameter x 33 in. high Surface Laboratory is mounted on top of the shear web and fastened to the three structural beams. Leveling of the Surface Laboratory is accomplished by the use of three auxiliary leveling legs that are operated after landing. The leveling legs are adjusted individually so that the Surface Laboratory and the shear web plane is level and perpendicular to the local vertical. The general arrangement of the legged lander is shown in Figure 4.2-21.

Estimated Weight

Touchdown Weight	2260 lb
Structures Weight	400 1ъ
Surface Laboratory	1270 1ь

4.2.3.2.2 <u>Pendulum Configuration</u> - The Pendulum concept grew from the need for a more stable, less complex legged landing system. A legged system was needed that could tolerate higher horizontal and vertical velocity components and stay upright on steeper slopes than the conventional legged lander with tripod struts and actuators in each strut. This was accomplished by the Pendulum because the overturning moments upon impact of the pads are not transferred thru the ball swivel to the payload. In addition, the free-hanging Surface Laboratory's gravity-leveled without the need of additional mechanism. Models were built and tested and the concept proved sound.

The Pendulum Lander is basically a bell-shaped Surface Laboratory supported by a ball joint at the junction of three equally spaced hollow trusses. The trusses provide a housing for the retraction mechanisms for the three legs and also support a 48 inch diameter ring undercarriage. The three legs are restrained from spreading on impact by six additional supports which are hinged to the undercarriage ring beneath the Surface Laboratory.

The Pendulum configuration has the advantage of a central impact attenuator in the pendulum ball joint which minimizes impact attenuator weight and simplifies

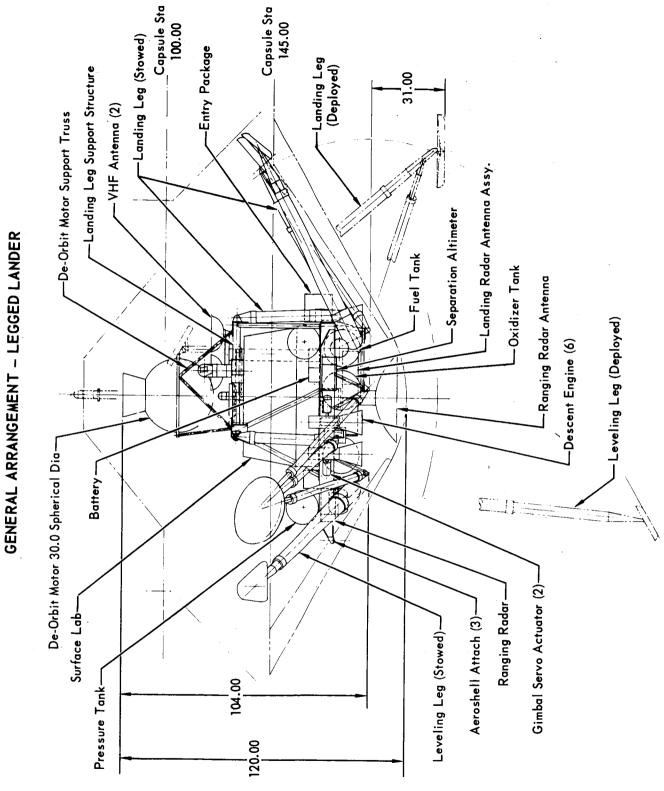


Figure 4.2-21

design. This configuration also provides gravity leveling of the Surface Laboratory up to a 30 degrees maximum slope. Another feature is the weight advantage gained by separating the pallet before impact.

The propulsion subsystem, including two fuel tanks, two oxidizer tanks, two pressure tanks, six descent engines and associated hardware, is mounted on a trussed framework (pallet) which bolts onto the top of the pendulum support structure and is pyrotechnically separated from the Capsule Lander at an altitude of ten feet above the Martian surface. The descent engines continue firing until the pallet is carried some predetermined safe distance away from the landing site.

The total weight of the Pendulum Lander before pallet separation is approximately 2260 lb. The touchdown weight is 1668 lb with a Surface Laboratory weight of 1266 lb. The Surface Laboratory contains 41.5 cubic feet with a surface area of 15.5 square feet at an angle of 15 degrees to the verticle and an additional 11.4 square feet at 50 degrees to the vertical.

The landing radar antenna mounted on the undercarriage of the pendulum support structure makes experiment deployment through the bottom of the Surface Laboratory impossible. The basic shape of the Surface Laboratory (truncated cone on a spherical segment) does not provide for efficient equipment packaging. Usable volume and radiation surface area of the Surface Laboratory are marginal for the 1979 mission and provide no growth potential. The general arrangement of the Pendulum Lander is shown in Figure 4.2-22.

Estimated Weight

Touchdown Weight	1668 lb
Structure Weight	356 1ъ
Surface Laboratory	1266 1ъ

4.2.3.2.3 <u>Triangle Configuration</u> - The Triangle Lander is a design which utilizes the relatively widespread footprint and proven landing technique of a Surveyor-type legged system but also incorporates a tumbling and recovery capability to further enhance mission success. The basic geometry of the system provides ample top and bottom laboratory surface area for deployed experiments access and minimum interference with view angles or operational envelopes. The geometry also lends itself to rather simple and direct mechanical deployment methods for the surface and sampling experiments.

The Capsule Lander consists basically of an equilateral triangle in planform with a side length of 115 inches. Depth of the lander is a constant 28 inches for the hexagon planform shape within the triangle. Outboard of the hex, the bottom of the

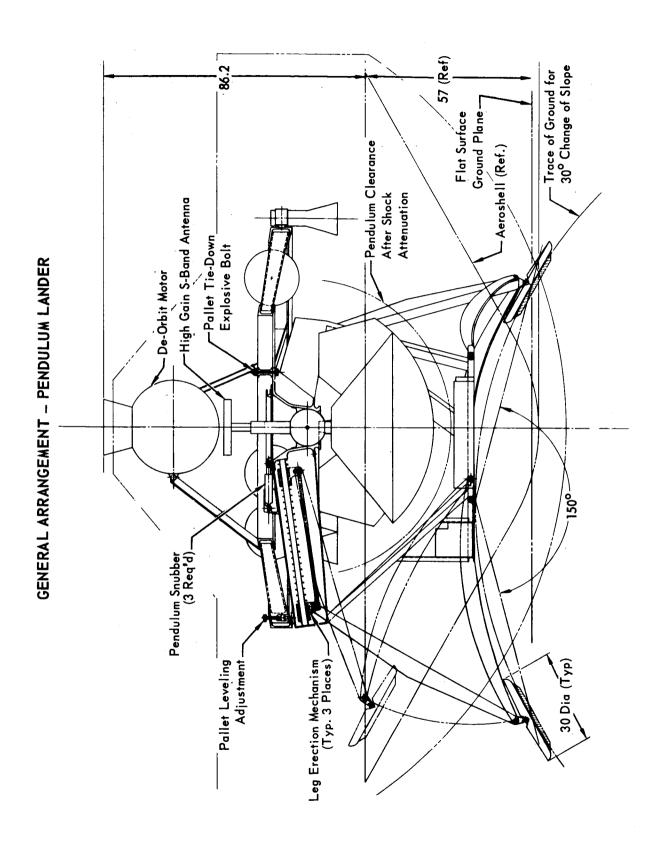


Figure 4.2-22

lander is tapered upward so that at the apex of the triangle the depth is 19 in. Each triangle tip houses a fuel, oxidizer, and pressurant tank for terminal propulsion as well as a primary shock attenuation cylinder for its landing leg. Each landing leg hinges at the attenuation cylinder and folds inboard. Lateral shock for each landing leg is attenuated by two struts and cylinders hinged at the bottom points of the hex and attached to a sliding sleeve on the leg. The sleeve is held in position by a ratchet lock when the leg is extended. The legs are of sufficient length to accommodate landing on a 30 degree slope change and still provide clearance for equipment.

The Surface Laboratory itself is contained within the hexagonal center section and provides approximately 50 cu. ft. of volume. Carry-thru structure is necessary in the lab to provide prime load paths and mounting structure. The three vertical sides of the lander will provide area for the radiating surface. Approximately 37.5 sq. ft. is available after deletion of areas shadowed by descent engines and bumpers. However, only 17.5 sq. ft. would be available as part of the lab; the other required areas would be adjacent to and extending into the tapered sections of the Bus. Laboratory equipment mounted or extended from the bottom of the lander include the surface sampler, alpha spectrometer, and subsurface probe. Deployment is relatively simple and there is adequate room for equipment operation.

Other bottom mounted equipment includes the landing radar antenna located approximately in the center of the triangle and rotated 35 degrees to the X-axis. Directly behind the antenna and below the bottom surface of the Laboratory are located electrical, guidance and control electronics, and telecommunication equipment. Adjacent to and on either side of the landing radar antenna are more guidance and control electronics, the altimeter antenna, and the ranging radar and separation altimeter which are used in conjunction with antennas on the Aeroshell. The Entry Science Package with vidicon cameras is mounted on the bottom along one edge of the triangle with view angles directed downward parallel to the Z-axis. It is apparent that the landing pads in the stowed position all but block out the landing radar so that the landing struts will have to be deployed immediately after Aeroshell separation in order for the radar to perform its functions.

The six terminal propulsion descent engines are located in pairs outboard of the basic triangular shape and centered midway along the sides. In the side elevation, the engines are located so that the exit nozzles extend below the bottom of the laboratory. The nozzles are scarfed and in close proximity to the Aeroshell inside surface which would be desirable when firing thru blow-out-ports in the

Aeroshell. The engine mounting radius of 40 inch about the Z-axis will provide good moment axes for control dynamics and the general engine location offers good versatility as to gimballing, canting, heat shielding, etc. Each pair of engines will be supplied with their particular set of propellant tanks located in one of the triangle points to avoid unbalanced flow.

In the event of tumbling after initial lander touchdown, each pair of descent engines are protected from impact and possible damage to the Laboratory by a surrounding truss structure and crushable bumper pad. Similar bumper pads are located on the ends and at the middle of the roll-over/leveling arms and surrounding the apexes of the Bus structure. These pads protect the Laboratory during tumbling and the top surface if the Bus comes to rest in a completely inverted position. On any but a flat surface with minor protuberances, the upper pads offer little protection from hard structure contact with the Martian surface.

Three roll-over/leveling arms are located along the upper periphery of the lander and are used to right the vehicle should it come to rest inverted or on any one of its three sides. Once righted, the arms are driven as required through angles of up to 270° to lift the lander off the surface and level the vehicle. In a landed attitude with an apex of the triangular shape pointing directly downhill, leveling on slopes up to 30° can be accomplished (assuming uphill landing struts are uncompressed). However, with an apex pointing uphill, leveling is only possible on slopes up to 18.5°. (uphill landing struts uncompressed), or 21° (both uphill struts compressed). One solution to providing 30° slope leveling capability regardless of landed attitude would be to telescope the leveling arms. Other solutions such as jettisoning the lander struts to lower the uphill portion of the lander might prove more desirable as this would also lower the vehicle, e.g., spread the footprint, and make the Laboratory less susceptible to vibration or overturn from high or gusting winds.

Equipment mounted on the top surface of the Laboratory is positioned to provide the required view angle and clearance with other equipment and structure. Repositioning or addition of new or different equipment will be possible due to the ample area available. Erectables include the high gain S-band antenna, visual imaging camera, and the atmospheric package and spectro-radiometer. The S-band omni-antenna and two VHF antennas are boxed as are the four in-situ mortar tubes. The mortar tubes are loaded so that in-situ experiment packages will be launched in four directions from the upper edges of the Laboratory with least possibility of entanglement of the data link cables.

The spherical solid propellant retrorocket, which is jettisoned immediately after retro-fire, is shown mounted by trusses to the integral Laboratory carry-thru structure. Both the spend rocket and its truss-work are jettisoned.

Lander mounting structure to the Aeroshell is not shown but could be accomplished at three points on the lander bottom surface near the apexes of the triangle. A more probable location would be on the bottom surface near the points of the enclosed hexagon to give six support points and stiffer arrangement. For continuity, the adapter support struts would probably be tied into the six lander support points. The general arrangement of the Triangular Lander is shown in Figure 4.2-23.

Estimated Weight

Touchdown Weight	2260 1ь
Impact System	477 1b
Surface Laboratory	1193 1b

4.2.3.2.4 Omni-Directional Torus Configuration - The Torus Lander configuration was developed to answer the need for a truly omni-directional non-mechanical impacting system. It represents the most efficient, well distributed load carrying structure and tolerates the greatest range in surface slopes, roughness, and density of any system studied. The Torus also has a unique gimbal-mounted payload which permits gravity leveling. By using the double torus design, and deflation of the upper torus after landing, greater view angles and radiation exposure are obtained.

The Torus Lander consists of a trunnion - supported 74 inch diameter payload; the payload is mounted to an inner gimbal ring which, in turn, is mounted to another gimbal ring to which the torus impact system is attached.

The 74 inch diameter payload structural ring is in the main support ring to which the Surface Laboratory is attached by a continuous structural attachment. Utilizing a group of six structural brackets, this ring also supports a lower ring to which the other systems equipments are attached.

The Surface Laboratory upper section is inclined 15° to the vertical and is used to provide the radiating area required. The Surface Laboratory extends down to the lower structural ring in the three pockets not occupied by the descent engines. The inner section of the Surface Laboratory has an open cylindrical section which is utilized to locate the fuel and oxidizer bottles. The lowering of the Surface Laboratory between the engines and the location of the fuel and oxidizer in the center of the Surface Laboratory provides a landed c.g. as low as possible.

The Aeroshell attachment of the lander is achieved by a continuous attachment around the 74 inch diameter payload structural ring.

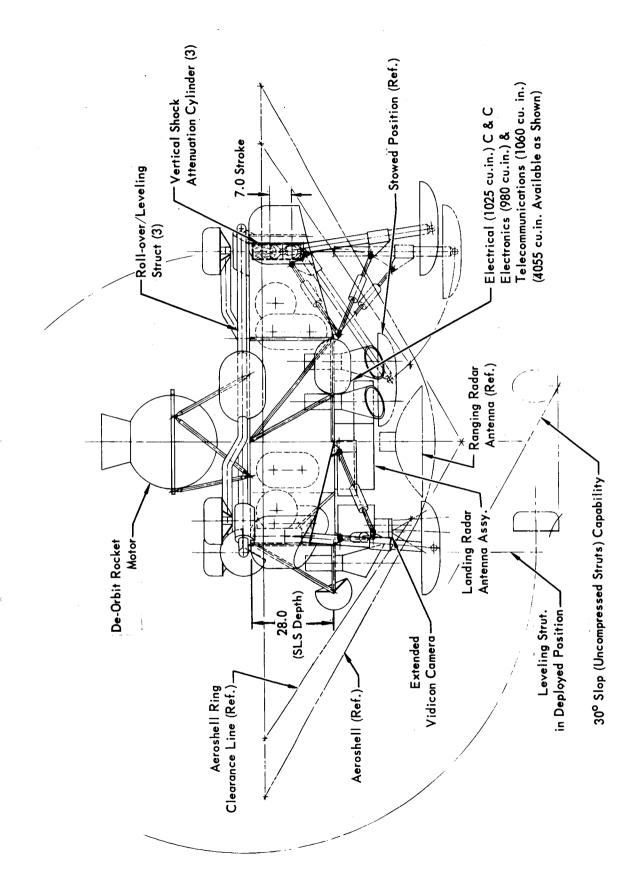


Figure 4.2-23

4-61

The landing system will be a double torus with a cross sectional diagonal of 48 inches. A 23 inch diameter outer torus is provided to assure that the lander will not come to rest on the edge of the torus. After the lander has come to rest, and the payload has been leveled, the upper torus and the outer torus will be deflated so experiment equipment may be deployed. The general arrangement of the Torus Lander is shown in Figure 4.2-24.

Estimated Weight

Touchdown Weight 777 1b
Total Torus Impact System 2260 1b
Surface Laboratory 893 1b

4.2.3.2.5 Mechanical Omni-Directional Configuration - The Mechanical Omni-Directional Lander is an approach to a true omni-directional landing with a simplified impact system, capable of righting and leveling after tumbling during landing. Impact energy is dissipated by crushing a foam material contained in fabric-covered pads at the top and bottom of the lander. Six legs, whose pads cover the top of the Surface Laboratory during impact, are rotated to erect and level the lander.

After impacting and coming to rest, the lander will be resting on its bottom or side. In either event, the setup procedures are identical:

- The three "short" legs are rotated 105° by pneumatic actuators.
- Using "harmonic drive" electric motors, the three "long" legs are rotated 8° and extended to their full length. Rotation then continues to a maximum of 191° if necessary for righting and leveling.

The leg and pad geometry of this lander are such that it can land and level on a 30° slope as well as clear 10 cm rocks and a 30° abrupt change of slope. With the rotation of the legs away from their impact position, the Surface Laboratory has an unrestricted view of Mars surface.

The Surface Laboratory has a volume of approximately 40 ft 3 and a radiator surface area of 22 ft 2 at an angle of 15° to the local vertical.

With pads sized for an impact velocity of 16 fps, the overall lander height is 80 inch with a maximum diameter of 103 inch. The lower pad has local reliefs for legs, descent engines, and landing radar. The upper pads form a six segment dome over the top surface of the Laboratory.

Three fuel tanks, three oxidizer tanks, six descent engines, and landing radar plus related electronics are shrouded by the lower pad below the Surface Laboratory. Three pressure tanks for terminal descent are mounted along the lower sides of the

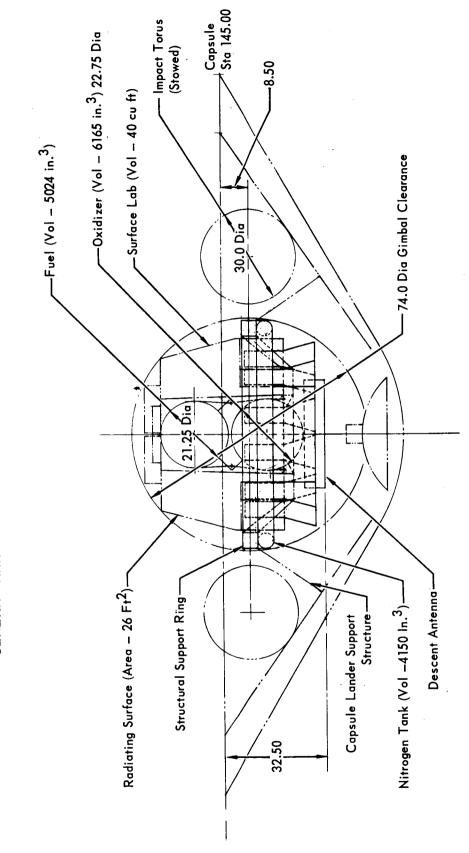


Figure 4.2-24

Laboratory. The general arrangement of the Mechanical Omni-Directional Lander is shown in Figure 4.2-25.

Estimated Weight

Touchdown Weight 2260 1b

Total Impact System 500 - 550 1b

Surface Laboratory 1100 - 1150 1b

4.2.3.2.6 <u>Lander Rover Configuration</u> - The Rover configuration evolved to satisfy four purposes: (1) compensate for landing error, (2) provide data over a wider area, (3) follow-up on trends or suggestive data, and (4) eliminate the lander propulsion disturbance/contamination.

The Rover Lander consists of a central package 61 inch x 61 inch wide x 80 inch long suspended between two tori. Each torus has an overall diameter of 112 inch. The 40 inch diameter cross section torus mounted on a 16 inch diameter wheel. The vehicle wheelbase is 118 inches.

The basic lander structure is a truss-frame 26 inches wide on center and 25 inches deep on center. The longitudinal members are 2.25 inch diameter $\times .050$ inch wall. Aluminum (7075-T6) tubing; cross-bracing, etc. is formed of 1.50 inch outside diameter $\times .032$ inch wall (7075-T6) aluminum tubing. At each end of the truss-frame is a torus support and an actuation and support unit.

The Surface Laboratory rests on top of the frame and overhangs each side-saddle style. The top surface of the Laboratory measures 61 inches x 54 inches long, giving adequate exposed area for deploying antennas, cameras, etc. The two end-faces of the Laboratory, excluding the bottom 6 inch, make up the radiation surfaces. Each radiation surface measures 31 inch x 61 inch and is tilted upward at 15°. The 6 inch wide strip of the Laboratory below radiator on each side, as well as approximately 3 sq. ft. of the bottom of the Laboratory on each end, is available for deploying experiments.

The propulsion engines are attached to the frame in pairs, 120° apart at 27 inches from the vertical of the lander. Fuel (3), oxidizer (2), and pressurant (2) bottles are spaced inside the frame and between the engines.

The landing electronics (i.e., antennas, electronics packages, batteries, etc.) are placed on the lower side of the frame inside the engine mounting circle.

The tori are designed to absorb the landing impact with no additional attenuation requirements. The design stroke requirement is .75 ft. The vehicle is also capable of sustaining a landing greater than 30° off-horizontal. The

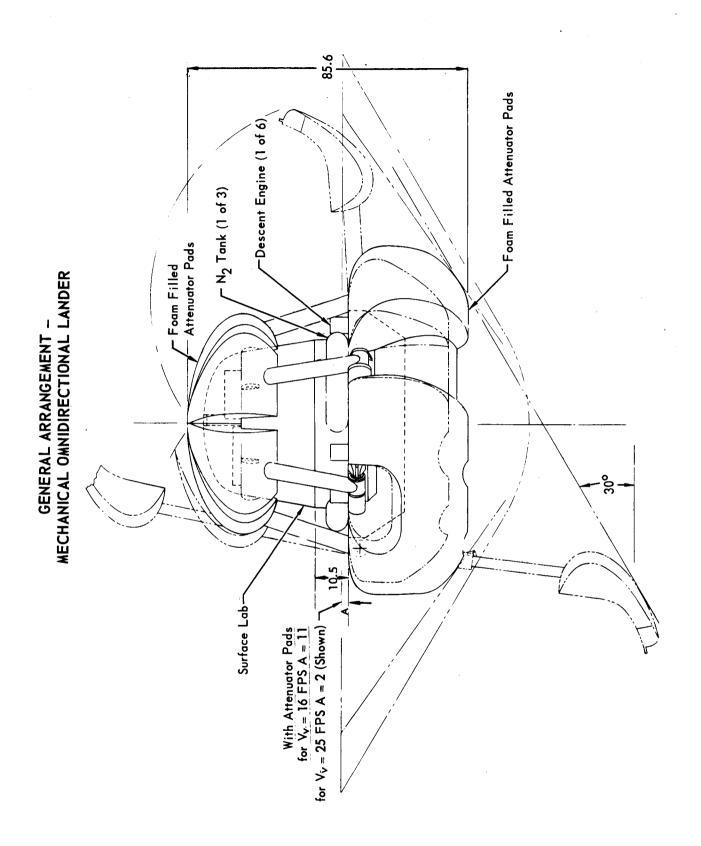


Figure 4.2-25

4-65

vehicle at rest will straddle a 30° hip-roof-shaped projection with 7.5 inches clearance in the middle.

Righting balloons are mounted on the outside hub of each torus. If the vehicle should come to rest on the side of one torus when landed, the balloon can be inflated (approximately 1 psi) causing the vehicle to right itself.

Vehicle motion after impact is controlled by a brake and gear motor acting on each torus, and a braking arm extending yoke-fashion outward from the lander. The braking arm can also be actuated by a gear motor.

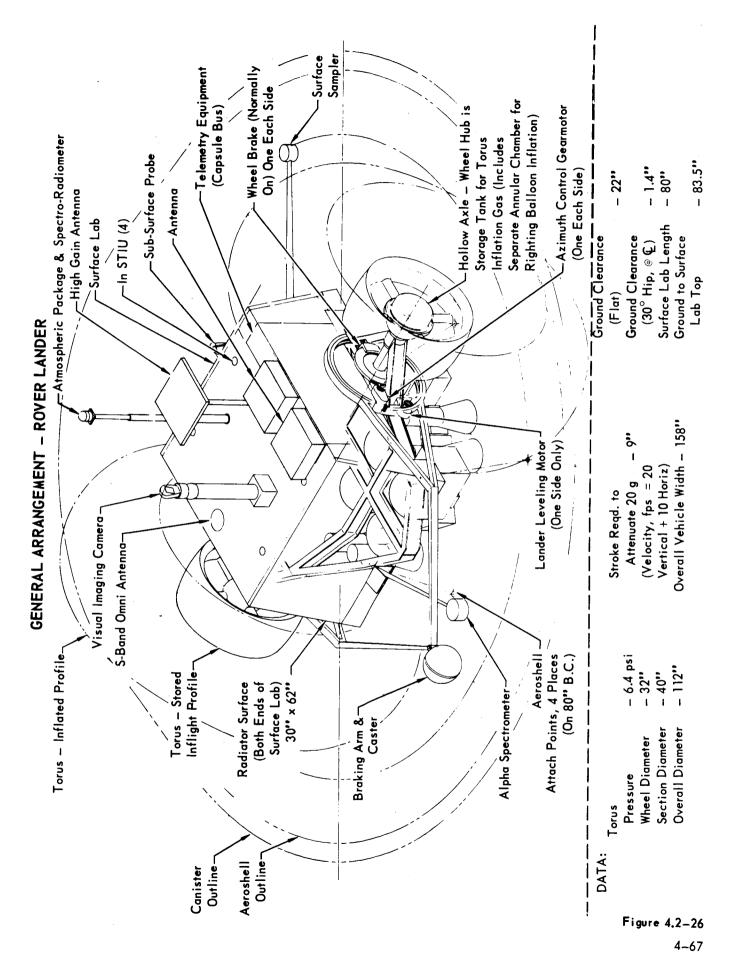
To orient the vehicle after landing, the gear motor in one torus is actuated, causing the vehicle to rotate, until the vehicle is perpendicular to the direction of slope. The braking arm gear motor is then actuated, rotating the Surface Laboratory into a level position.

The Rover Vehicle can also travel across the Martian surface, using the two gear-driven tori. It can travel in a straight line, or in a circle, turning if and when desired. Travel is limited only by the size of the motors, the capacity of the batteries, and the nature of the Martian surface. The general arragnement of the Rover Lander is shown in Figure 4.2-26.

Estimated Weight

Touchdown Weight	2260 1ъ
Structure Weight	538 1ъ
Surface Laboratory	1132 1b

- 4.2.3.2.7 <u>Alternate Lander Concepts</u> Nine alternate lander concepts were evaluated in the following eight areas of comparison before being rejected. This evaluation is shown in Figure 4.2-27.
 - Effect on experiments and thermal control.
 - Weight (efficiency of design)
 - Effect on sterilization
 - Complexity of mechanisms (reliability)
 - Susceptability to damage (landing system and payload function after impact)
 - Omni-directional ability, stability, footprint.
 - Effect on entry cone/lander separation.
 - Effect on entry cone (stowage, c.g. location).
- 4.2.3.3 <u>Evaluation</u> The evaluation of the six more promising candidates is summarized in Figure 4.2-28. Eight major selection criteria were used to evaluate the candidates. Each of the eight criteria was further divided into subfactors.



ALTERNATE LANDER CONCEPTS

CONFIGURATION	DESCRIPTION
SPHERICAL MECHANICAL LEGGED OMNIDIRECTIONAL LANDER	Consists of a spherical payload surrounded by a layer of low energy absorption material such as foamed plastic. Telescoping flip-out paddles are deployed by cables wrapping on motor driven drums at the leg pivot points. When the paddles are in position the motors reverse and engage clutches to drive the legs. This erects the Lander. The legs are then driven separately to level the Lander. The top hemisphere of attenuator material is doffed to expose radiators and experiments prior to their operation.
SPHERICAL PNEUMATIC LEGGED OMNIDIRECTIONAL LANDER	Consists of spherical payload surrounded by a layer of low energy absorption material such as foamed plastic. An inflatable tripod wrapped around the outside of the Lander is filled after landing to right and level and payload. The top hemisphere is doffed to expose radiators and experiments prior to their operation.
MULTIPLE TORUS FIXED PAYLOAD OMNIDIRECTIONAL LANDER	An omni-directional system which employs three or four small tori similar to balloon tires extending radially on arms from the payload support structure. After coming to rest an internal instrument senses which side is up and deploys the proper one of the dual antennas or equipment which must be up. Any leveling required is accomplished by the individual equipment. Initial impact is on the "bottom side" i.e., the side containing the deceleration thrusters.

ADVANTAGES

1. a) Diffic b) Anten

2. a) Ineffi b) Bulky 3. Steriliza cult beca 4. Poor reli before of 5. Mechanic

1. a) Diffic

b) Anten

c) Lower

a) Ineffic

b) Bulky

Steriliza cult bec Poor rel before o

1. Duplicat

2. a) Undes

3. a) Compl

5. Small foo

6. Makes fr 7. a) Stowa b) Cente

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b) Each the to

b) Increa 4. a) Possi

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of equ

c) Lower

۱.	Offers	some	protection	from dust	and	erosion	bу	windblown particles.	
2.	a) Omn	i-dire	ctional						

2. Offers some protection from dust and erosion by windblown particles.

1. a) Greater payload exposure compared to the single large torus system.

2. a) Overall reduction in weight due to elimination of the gimbal rings

4. Leveling of individual equipment is accomplished by gravity with only

5. Moderately low center of gravity when stowed in entry cone.

b) Reduced inflation volume from single torus design.

3. Easy separation from entry cone

1. Reliable attenuation system free of mechanisms.

4. Moderately low center of gravity in entry cone.

b) Stability on impact not required.

4. Easy separation from entry cone.

b) Simple equipment installation.

lock and unlock functions required.

b) Stability on impact not required.

5. a) No mechanical arms to jam during impact.

3. a) Omni-directional

c) Large footprint

3. Easily sterilizable.

6. a) Omni-directional

b) Stability on impact not required

c) Large footprint

DISADVANTAGES

olt equipment packing

na installations become very difficult especially if the crushable

shell is not R.F. transparent.

half of payload is covered by attenuator greatly limiting exposure ipment.

cient use of attenuator material.

and heavy design
ion is difficult. Heat penetration through the outer cover is diffi-

use this material is an insulator.

ability because of the many functions which must be successful veration of equipment and experiments.

ılt equipment packing

ia installations become very difficult especially if the crushable ishell is not R.F. transparent.

half of payload is covered.

al legs must function after impact.

ient use of attenuator material. and heavy design.

and heavy design. tion is difficult. Heat penetration through the outer cover is diffi-

use this material is an insulator.

ability because of the many functions which must be successful peration of equipment and experiments.

on of some equipment.

rable load paths for flat-side landing conditions.

orus and set of arms must be capable of taking from 50% to 75% of al impact loads.

x torus fill system

sed complexity inside payload.

illity of puncture or slow leak caused by an unanticipated sharp ob-This problem can be virtually eliminated by a lightweight small inner

of mylar which could support the light Martian weight of the Lander.

ent exit from entry cone impractical.

e in entry cone looks difficult.

of gravity in stowed position high in entry cone.

ALTERNATE LANDER CONCEPTS (Continued)

CONFIGURATION	DESCRIPTION
MULTIPLE TORUS GIMBALLED PAYLOAD OMNIDIRECTIONAL LANDER	An omni-directional system which employs three or four smaller tori, similar to balloon tires, extending radially on arms from the payload support structure. This system has a gimbaled payload. Initial impact is on the "bottom side" i.e., the side which contains the deceleration thrusters.
SINGLE TORUS FIXED PAYLOAD OMNIDIRECTIONAL LANDER	Consists of a payload package rigidly attached in the center of a torus shaped impact bag with no provisions for having a particular side up after impact. Initial impact is, however, on the "bottom side" i.e., the side containing the deceleration thrusters. After coming to rest an internal instrument senses which side is up and deploys the proper one of the dual antennas or equipment which must be up. Any leveling required is accomplished by the individual equipment. Bag pressure need only be approximately 1.0 psi.
SINGLE TORUS GIMBALLED PAYLOAD OMNIDIRECTIONAL LANDER	Consists of a central payload package with a low off-centered c.g. The payload is trunion mounted in a gimbal ring which is in turn trunion mounted in an outer ring to which the inflatable torus bag is attached. The lander is stowed deflated with gimbal rings and payload locked against rotation. Following separation, inflation, and impact the gimbal rings and payload are unlocked. Initial impact is on the "bottom side" i.e., the side containing the deceleration thrusters. After the vehicle has come to rest, the gimbals are unlocked and the payload is allowed to swivel to the level up-right position. When this has been accomplished, the trunions are locked. The vehicle is now ready for operation. Bag pressure need only be approximately 1.0 psi.

Figure 4.2–27 (Continued)

4-69 -1

ADVANTAGES

1. a) Overall sy

2. Complex torus

a) Equipment

1. a) Large tori

2. a) This desi

b) Duplication

b) Increased 3. Possibility o

This problem

a) A lightwe

light Martian

b) Radial we

1. Large torus s

2. Short fill time

3. a) Possibility

This problem

1) A lightweig

Radial web

b) Failure to

light Martic

of which co

from Mars :

two of wh

operations

b) Possibility

jest. This

inner blad Lander. c) Failure to mission from 4. Small footpris 5. Front exit fro 6. a) Stowage ir b) Center of 9

b) Undesirab

c) Each torus

the total in

- 1. a) Greater payload exposure compared to the single large torus. b) Simple equipment installation.
- 2. Light weight inflation system (smaller gas volume than single torus).
- 3. Easily sterilizable. 4. Righting and leveling are accomplished by gravity with only lock and
- unlock functions required.

- No mechanical arms to jam during impact.

- 6. a) Omni-directional
- - b) Stability on impact not required.

- 1. Overall reduction in weight due to elimination of the gimbal rings and
- locks. 2. Easily sterilizable
- 3. Leveling is accomplished by gravity with only lock and unlock functions
- 4. Failure to level would result in only a reduced mission. 5. a) Omni-directional
 - b) Stability on impact not required.
 - c) Large footprint area makes this one of the most tolerant, of all sys-
 - tems considered, to surface density and texture conditions.
- 6. Easy separation from entry cone. 7. Low center of gravity when stowed in entry cone.
- 1. Simple equipment installation
- 2. Easily sterilizable
- 3. Righting and leveling are accomplished by gravity with only lock and unlock functions required.
- 4. a) Omni-directional

 - b) Stability on impact not required.
- 5. Easy separation from entry cone.
- c) Large footprint area makes this one of the most tolerant of all systems considered to surface density and texture conditions.
- 6. Low center of gravity when stowed in entry cone.
 - - - - 4-69-2

DISADVANTAGES

tem is heavy.

load paths for flat-side landing conditions. and set of arms must be capable of taking from 50% to 75% of

pact loads.

fill system.

exposed to dust and erosion from windblown particles.

of puncture or slow leak caused by an unanticipated sharp ob-

problem can be virtually eliminated by a lightweight smaller ler of mylar which could support the light Martian weight of the

right from the inverted position could result in no data trans-

m Mars surface. t area

m entry cone impractical.

entry cone looks difficult. ravity in stowed position high in entry cone.

s shape may hamper radiation efficiency.

n of some equipment.

n requires the torus fill very quickly just prior to impact. complexity inside payload.

puncture or slow leak caused by an unanticipated sharp object.

can be virtually eliminated by several methods:

ght smaller inner bladder of mylar could be used to support the

weight of the Lander.

s which break the torus into six or eight segments any one or ch could be punctured without hompering the post landing

ape may hamper radiation efficiency.

of large volume just prior to impact.

of puncture or slow leak caused by an unanticipated sharp object.

can be virtually eliminated by several methods:

ht smaller inner bladder of mylar could be used to support the n weight of the Lander.

s which break the torus into six or eight segments any one or two uld be punctured without hampering the post landing operations.

ight from an inverted position could result in no data transmission urface.

ALTERNATE LANDER CONCEPTS (Continued)

CONFIGURATION	DESCRIPTION
TRIPLE TORUS FIXED PAYLOAD OMNIDIRECTIONAL LANDER	Differs from single torus design as follows: a) There are three independent tori two of which are designed and sized to take the impact loads and support the payload. The third torus, wh requires just a fraction of a psi pressure, guards against the possibility of coming to rest on edge. b) The two large tori have bonded in radial bungee cords which allow for retracting whichever torus is on top after landing. This allows greater free exposure of the payload. Main bag pressure need only be approximately 1.0 psi.
ROCKET STABILIZED LEGGED LANDER	A legged lander with an attempt made to prevent rebound by mounting upward firing rockets to each landing pad. The rockets are set to fire on initial contact of each pad.
BASE PAD LEGGED LANDER	Employs a single large crushable pad directly under the payload. The pad absorbs the impact energy. The vehicle is kept up-right by a number (say 4 or 5) of relatively lightweight deflatable out-rigger arms working with an attitude rocket control system. After coming to rest, the arms are used to level the payload.

Figure 4.2-27 (Continued)

ADVANTAGES

- 1. a) Good experiment and radiator exposure.
- b) Simple equipment installations.
- 2. a) Lighter than single torus design.
- b) Overall reduction in weight due to elimination of the gimbal rings and locks. Easily sterilizable.
- 4. a) Leveling is accomplished by gravity with only lock and unlock functions required.
- b) Smaller inflated volume than single torus (requires a smaller gas fill system).
- 5. a) No mechanical arms to jam during impact.
 - b) Low gas pressure, easy to seal.
 - c) Failure to level would result in only a reduced mission.
- 6. a) Omni-directional b) Stability on impact not required.
 - c) Large footprint area makes this one of the most tolerant of all
- systems considered to surface density and texture conditions. 7. Easy separation from entry cone.
- 8. Low center of gravity when stowed in entry cone.
- 1. Experiments and radiators almost completely exposed.
- 2. Easily sterilizable.
- 3. Simple gear geometry which tends toward greater reliability. 4. Failure of one leg or of the leveling system would result in a reduced

- 5. Should stay up-right when impacting on slopes.
- mission but would not be catastrophic.

1. Good exposure of experiments and radiators.

3. Easy sterilizable.

Stowable in entry cone.

- - - - b) Payload expos c) Components of landing impact
 - 5. a) Lacks omni-di b) Changing leg
 - c) Small landing

1. Duplication of so

a) Short fill times b) Additional man

single torus de

object. This p 1) A lightweig

the light Mo 2) Radial web

one or two

landing ope b) Equipment exp

c) Increased comp

3. a) Possibility of

1. Possible fogging

2. Excess weight:

a) Each leg must

b) Each strut has

c) Three or four r

b) Complex level 4. a) Any delay in p

b) Rebound ener

3. a) Complex contr-

- 6. Makes front entry 7. a) Possible stow b) Center of gray
- 1. Bottom of paylor installation of th 2. Moderately efficient use of shock attenuating material.
- questionable. 4. Failure of a leg or of the leveling system would result in a reduced 2. Complex levelin
 - arms are used. mission but would not be catastrophic. 3. Payload exposed
- 5. Increased footprint area over most other legged landers. 4. a) Lacks omni-d
 - aggravate the 5. Makes front exit 6. Center of gravit

DISADVANTAGES

me equipment.

for large volumes just prior to impact.
ifolding and valving in fill and deflate system over sign.

plexity inside payload.

puncture of slow leak caused by an unanticipated sharp roblem can be virtually eliminated by several methods:

ht smaller inner bladder of mylar could be used to support irtian weight of the lander.

s which break the torus into six or eight segments any of which could be punctured without hampering the post

erations.
Hosed to windblown dust and erosion.

of instruments and equipment from upward firing rockets.

be designed to withstand 50% to 75% of landing load. an attenuator.

ockets and controls (one for each pad).

ols for split second timing of rocket firing on impact. ing mechanism required after impact.

ad rocket firing may greatly increase landing shock.

ed to dust and erosion from windblown particles.

legs must function for leveling after being subjected to bloads.

rectional capabilities.

geometry during impact reduces stability.

pad area limits surface density tolerance.

, from entry cone impractical.

age problem presented by additional rockets.

ity in stowed position raised by rocket mass.

Id completely covered with shock absorber pad making

ne downward looking Doppler Radar Antenna very

g mechanism, increased complexity if more than three

I to dust erosion from windblown particles.

irectional capabilities.

gy stored when outriggers deflect. This should instability on slopes or during side drift landing.

from entry cone impractical.

while stowed higher than most other legged landers.

LANDER CONFIGURATION COMPARISON

CANDIDATE LANDER	ADVANTAGES	DISADVANTAGES
Torus	 a) Omnidirectional b) Stability on Impact Not Required c) Large Footprint Area Gravity Leveling Simple Impact Attenuator Minimum Development Time 	 Poor Storage Life Marginal Environmental Control Marginal for Deployed Experiments Limited for Growth Not Mobil RTG Cannot be Installed LowProbability of IDP Installation Marginal for Telecommunication Installation
Pendulum	 Most Stable of the Legged Systems Good Storage Life Gravity Leveling Simple Impact Attenuator Minimum Development Time Easy Fabrication and Checkout High Operational Reliability 	1. Low Environmental Control 2. Highly Complicated Deployed Experiment 3. Not Mobil 4. Low Probability of RTG Installation 5. Low Probability of IDP Installation
Legged	 Good Storage Life Good Environmental Control Easy Deployment of Experiment Simple Gear Geometry 	1. Complex Leveling Mechanism 2. Lacks Omnidirectional Capabilities 3. Assembly and Checkout Difficult 4. Low Probability of IDP Installation 5. Not Stable on Slopes at Impact
Triangle	 Very Good Deployment of Experiments Very Good Telecommunication Installation Good Storage Life Minimum Complexity Experiments Good for RTG Installation Good Terminal Propulsion Installation 	1. Not Stable on Impact 2. Most Complicated Impact Attenuator 3. Marginal on Leveling Capability 4. Maximum Development Time 5. Low Probability of IDP 6. Fabrication Difficult
Mechanical Omnidirectional	 a) Omnidirectional b) Stability on Impact Not Required c) Large Footprint Area Simplest Impact Attenuator Good Storage Life 	1. Marginal for Telecommunication Installation 2. Marginal for Deployed Experiments 3. Long Development Time 4. No Growth Potential 5. Not Mobil 6. RTG Cannot be Installed 7. Low Probability of IDP Installation
Rover	 a) Omnidirectional b) Stability on Impact Not Required c) Large Footprint Area Simple Impact Attenuator Good Environmental System Minimum Complex Experiments Good Telecommunication Installation Good for RTG Installation Good Growth Potential Mobil 	Poor Storage Life Low Probability of IDP Installation Assembly and Checkout Difficult

A value of 1 through 10 was then assigned to each subfactor with 10 being the highest and given to the configuration which best meets the particular requirement. The scores were totaled for each criteria and a weighing factor was applied to derive the overall evaluation, as shown in Figure 4.2-29.

From this figure was derived the following:

- Omni-Directional Landers rank 1, 2, and 3 in landing success with a fairly good spread between these 3 and the others.
- Gravity leveling and simple noninflatable impact attenuation systems score best under operational reliability.
- The major differences between the upper and lower ranking landers under performance are their ability to meet environmental control requirements, noncomplicated experiment installation and good telecommunication antenna installation.
- Landers having noncomplicated experiment installations, and capability of varying the shape and volume rank the highest under Surface Laboratory optimization.
- The system compatibility evaluation indicates that, except for the terminal propulsion subsystem, the six lander configurations are about equal.
- Under development time and risk, inflatable omni-directional landers rank the highest. This analysis is widely understood and highly refined. Any development and test program would benefit from their durability (few test articles required) and the ease with which variations in performance are obtained.
- The three top ranking landers under flexibility have better growth capability, greater mobility or possibility of partial mobility, can accommodate an RTG, and have good field of view for the experiment installation.
- The cost evaluation indicates that ease of fabrication and development are the prime factors.

The results of the evaluation indicate that the landing success, reliability, performance, and development time and risk are the major factors in ranking the preferred concepts.

The Rover was number 1 in preference due to omni-directional landing capability, good performance, good growth potential and Surface Laboratory optimization.

The Torus ranked next because of high scores from a landing success and operational reliability standpoint but was unsuitable as a standard growth vehicle.

LANDER CONFIGURATION EVALUATION

		MISSION SUCCESS (.35) OPERATIONAL																					Р	ERI	FOR	MΑ
	L	.ANI	DIN	3 SU	CCE	:SS	(.18)						IAL (.17	7)		COM	1PLI	AN	CE (.07)					ACE FIMI:	
Candidate Sub Factors	Tumbling	Increased Landing Velocity	Increased Slope	Lower Bearing Capacity	Larger Rocks	Ridges	Total	Weighted	Sterilization Susceptibility	Storage Life	Impact Attenuator Complexity	Orientation Confidence Level	Total	Weighted	Environmental Control System	Experiments Complexity	Antenna Installation	Stability	Payload Weight Capability	Total	Weighted	Component Packaging	Experiment Installation	Electronic Installation	Environmental Control	S
Torus	10	10	10	10	10	10	60	18	9	3	7	8	27	. 13	6	4	4	9	3	26	04	7	3	5	8	1
Pendulum	4	5	4	3	3	4	23	07	10	10	6	10	36	.17	1	1	8	10	7	27	.04	6	1	3	1	1
Legged	1	1	1	1	4	2	10	03	10	8	3	1	22	.10	10	7	7	10	7	41	.07	8	8	3	9	1
Triangle	6	3	5	6	5	5	30) 09	8	8	1	3	20	.09	8	9	10	10	6	43	.07	8	10	10	6	
Mech OMNI	9	7	8	8	8	7	47	14	7	10	10	5	32	.15	5	3	1	10	3	22	04	7	5	7	8	1
Rover	10	9	9	9	10	9	56	17	9	3	8	9	29	.14	10	10	7	10	2	39	.06	8	10	9	10	1

Figure 4.2-29

4-73 -1

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-	CE				T									ELO Tik																
	3OR ION		ORY 7)			CO1		/STE					Α	ND				FL	EXI	BIL	ΙΤΥ	(.15)	ı		С	ost	(.10)			
h		, <u>.</u>	· /	т	_	T	1PA	LIBI	L!!	Y (.0)6) 		RISK	(.20	0)	L.,		_		1	_	,			· ·	Ţ		_		
	Shape	Volume	Total	Weighted	I.D.P. Installation	Aerodynamic Decelerator	Terminal Propulsion	Launch Vehicle Envelope	Aeroshell Separation	Total	Weighted	Time	Risk	Total	Weighted	Growth Capability	Mobility (Total/Partial)	RTG Installation	Extended Mission Capability	Experiment Flexibility	c.g. Location in Aeroshell	Total	Weighted	Accessibility Assembly and Checkout	Fabrication Ease	Development Testing	Total	Weighted	Weighted Score	Rank
			37	/						30				20	/							21	/				24	\overline{A}		
	2	2		04	ו	10	3	9	7	/.	05	10	10	/.	.20	2		2	3	3	10		.05	6	8	10		09	.77	2
			26	/						29	/			14	7							31					28	7		
,	3	2		.03	ı	10	2	9	7	/.	.05	9	5		.14	5	3	4	6	6	7		.08	10	10	8	/	10	.68	3
			52	/						36	/			8	/							50					10 /	7		
	7	7	/.	.05	1	9	10	9	7		06	7	1	/	.08	7	7	9	9	8	10		. 13	3	3	4	/	04	.56	6
			58							37				4								55					10	1		
,	8	8		.06	1	9	10	10	7	/.	06	1	3		.04	8	8	10	10	9	10		. 14	8	1	1). ا	04	.59	4
			39							25				9							-	17					11	7		
,	1	1	/.	04	1	9	1	7	7	<i>.</i> .	04	3	6	<i>/</i> .	.09	1	1	1	1	3	10		.04	4	4	3). J.	04	.58	5
			67	/						33				17								59					16	1		
	10	10	/.	07	١	8	8	9	7	<i>.</i>	05	8	9	<i>[</i> .	17	10	10	10	10	10	9		. 15	4	6	6	<i>.</i> .	06	.87	١

Both of these concepts use an inflatable fabric system and are capable of recovering from tumbling on landing. In as much as this is the only feasible omni-directional system, additional study of it is warranted. Because the rover is essentially a dual torus system with growth and mobility capability it is chosen as a configuration worthy of additional study.

The Pendulum Lander ranks third because of good landing success and high reliability but is very poor for Surface Laboratory optimization.

The Triangle Lander ranks fourth but, because of lack of detail mechanism design and some suspicion that the design is not adequate, this ranking was somewhat suspect. In particular the crushable bumper pads are considered inadequate on all but flat surfaces and the roll over/leveling arms are not capable of actually functioning. Both of these problems are capable of solution but the system will get much heavier. One configuration detail of the Triangular Lander is optimum - thin, low silhouette with maximum pitch and yaw moments of inertia - this promotes increased landing stability.

4.2.3.4 <u>Composite Legged Lander</u> - As a result of the study, a new legged lander configuration was derived, combining the desirable features found in the previous configurations. It is shown in Figure 4.2-30.

This lander is a large, squat, cylinder (24 inch high x 130 inch diameter) supported on three legs. It has a low c.g. location and a large pitch/yaw moment of inertia. Both these features contribute to increased stability.

The central core of the cylinder contains all the Capsule Bus equipment less the legs. From this core, structure extends to form three leg wells and mounting points for the Surface Laboratory. The core is 76 inches in diameter and 27 inches high. Interior equipment placement in the Capsule Bus is highly flexible. The Bus volume can accommodate a wide variety of descent engine and landing radar sizes and arrangements.

The Surface Laboratory is located outboard and around the Capsule Bus and between the gear wells. The Surface Laboratory sections are utilized through their mutual connection to a structural conduit tunnel running around their inner top edge.

The volume and radiator area of the Surface Laboratory totals 76.6 ft^3 and 526 ft^2 . These areas and volumes are higher than presently required but permit modularization and future growth.

Each leg consists of a round strut, which is ball mounted at the top and trunnion mounted near the middle. Between the trunnions and the tubular strut

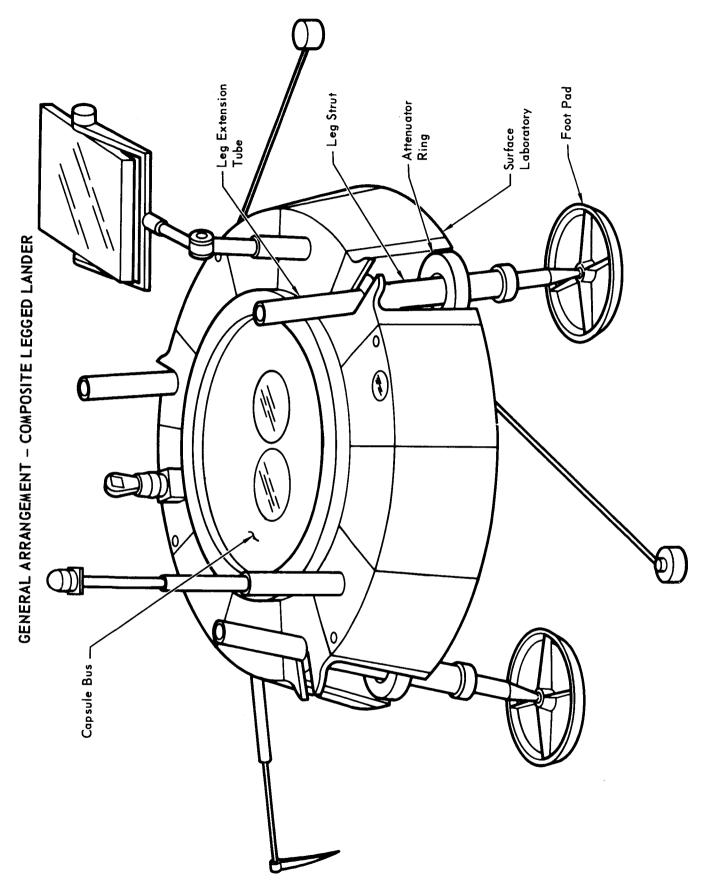


Figure 4.2-30

is a ring of crushable material that attenuates side loads. Attached to the structure that contains this ring of attenuator is a locking plate which is actuated after landing by the same screw jack that positions the leg. This locking plate eliminates side sway in the leg after landing. In the middle of the strut is a hollow threaded extension tube which can be extended to level the lander. An attenuator of crushable material is contained in this extension tube. This attenuator reduces axial strut landing loads. A 26 inch diameter footpad is ball socket mounted on the end of the extension tube.

A screw jack actuator is used to move each leg from the stowed position to the landing position and then back to the leveling position. Latches hold the leg in the landing position so impact loads are not carried through the screw jack. These latches have automatic engagement and pyrotechnic release.

The advantages of this configuration are:

- Low flat silhouette with maximum pitch and yaw moments of inertia.
- Low center of gravity and minimum c.g. height/gear span ratio.
- Large Surface Laboratory volume.
- Large unrestricted vertical surface area for radiators.
- Easy access for installation/maintenance.
- Surface Laboratory shape well suited to modularization.
- Leveling capability more easily incorporated in single strut gear.
- Short reach and minimum obstruction to soil sampler, probe, alphaspectrometer, etc.

The major disadvantage is its nontumbling limitation.

4.3 TERMINAL PROPULSION SELECTION TRADE STUDY - A terminal propulsion subsystem configuration is selected that meets the requirements of VOYAGER for decelerating the Capsule Bus to low velocity and to a controlled landing attitude. The selection was made in two steps from a broad spectrum of feasible configurations. The first step narrowed the field to the high value candidates by comparing each concept and applying certain selection factors. The second step refined the analysis of each of the candidates and compared them, using a set of criteria that is being used in all trade studies for the VOYAGER Phase B program.

The VOYAGER Capsule at entry consists of the Aeroshell, the Capsule Bus lander, the Surface Laboratory, and the Entry Science Package. The Aeroshell, although it produces 90 - 97% of the deceleration of the Flight Capsule, must be augmented by a terminal propulsion subsystem to accomplish soft landings (defined as $\rm V_V \leq 25$ ft/sec and $\rm V_h \leq 10$ ft/sec in Reference (a)). The terminal propulsion subsystem selected must shut down at preset conditions without imparting destablizing torques.

The scope of the trade study is limited to trades within the terminal propulsion subsystem itself and between it and the guidance and control mode of operation. The initiation conditions assumed for the first step of the comparison of 1, 3, 4, and 6-engine configurations were 600 ft/sec at 10,000 ft in an out-the-back of the Aeroshell (or fire-in-the-hole) separation technique. The type of tankage and the propellant distribution networks were kept identical in form between the several rocket configurations. Sizing of the tanks was dependent upon the total impulse required for each mode and the average specific impulse attainable from each engine and propellant combination. Thus, weight and volume differences were a function only of numbers of thrust chambers and their arrangement.

The engine selected will be combined with a deployable aerodynamic decelerator (Ballute or parachute) or will be used alone in an optimum deceleration configuration in a subsequent trade study (Section 4.5).

4.3.1 <u>Summary</u> - A multiple engine configuration consisting of four alternately canted engines, operated differentially to provide inherent pitch, yaw, and roll control torques, is selected for the Capsule Bus terminal propulsion subsystem. The engines will be used in a preprogrammed guidance mode in conjunction with a Lunar Module type of landing radar and a set of integrating rate gyroscopes and an axial accelerometer. The nominal terminal conditions at shutdown are set at a vertical descent velocity of 10 ft/sec and a zero horizontal velocity at 10 ft

- altitude. Selection of a differential drag method of Aeroshell/Capsule Lander separation would permit a reduction in total impulse requirement. The thrust chambers are sized for 650 lbs thrust (min.) continuously variable up to 6,500 lbs (maximum) for a peak thrust/weight ratio of 6.4:1 with part of this thrust used for differential attitude control.
- 4.3.2 Functional & Technical Requirements The terminal propulsion phase of the Capsule consists of either Aeroshell-Capsule Lander or aerodynamic decelerator-lander separation, orientation to decelerate the vehicle along its total velocity vector (which produces a gravity turn), and a final descent with the vehicle's attitude held fixed and at a constant velocity that established the landing conditions (see Figure 4.3-1). Separation of the lander from the Aeroshell is the subject of a separate trade study (Section 4.6) but will depend either on differential aerodynamic drag (preferred) or a high thrust level rocket firing (propulsion only). The velocity increment required for separation is a function of atmospheric density and wind speeds, possible interference between the landing radar and the Aeroshell, and possible recontact. Radar damage and beam propagation interferences are real possibilities with any propulsion system. The landing radar employed is a modified Lunar Module landing radar having a range beam and four doppler velocity beams. The severity of this radar interference must be minimized to produce high probability of mission success.

Orientation of the vehicle's thrust vector to the total velocity vector cancels the lateral ground velocity whether inertially or atmospherically induced. The gravity turn can result eventually in an alignment of the Capsule axis and the local vertical. In the low density atmospheres (VM-7 and VM-8) the propulsion subsystem must decelerate rapidly to provide adequate time for this gravity turn.

The landing conditions assumed have little direct effect on impulse requirements, and hence, deceleration optimization, because the final velocity is virtually zero. No programmed Capsule maneuver occurs at engine shutdown; the Capsule merely drops under Martian gravity to the surface.

4.3.2.1 Scientific Requirements - The deceleration subsystems facilitate the accumulation of entry science data by helping to provide a long, slow descent. The propulsion subsystem must have a minimal effect on sensor performance. The selected engine should not decrease materially the ability to reconstitute the atmosphere. Thus, off—c.g. location of the accelerometers and gyros is undesirable. There is also an operational requirement that the effect of the exhaust plumes of the

TERMINAL DESCENT SEQUENCE

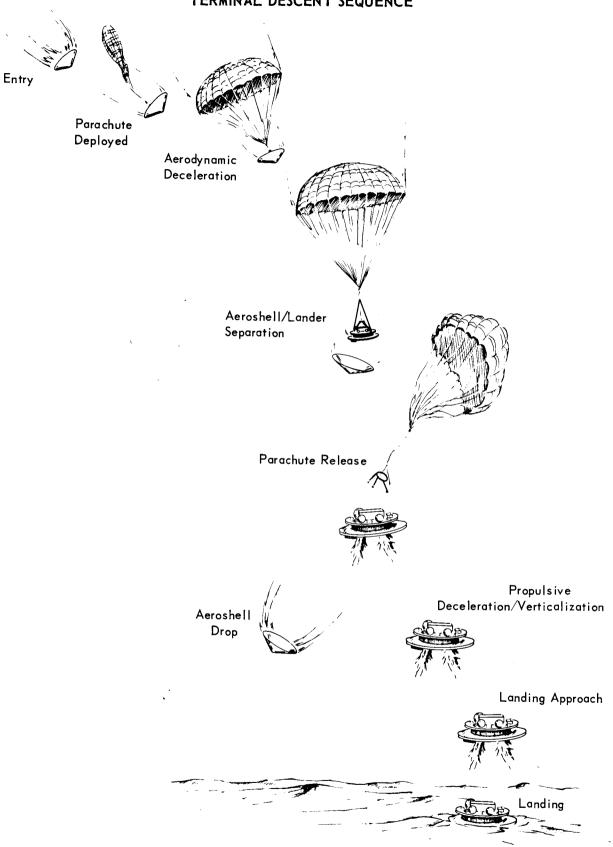


Figure 4.3-1

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rockets on the radar beams be minimized. Similarly, impingement of the exhaust on the Martian surface must not have deleterious effects on Capsule performance and should minimize contamination of the surface under and near the Capsule.

4.3.2.2 Design Requirements - The Capsule Lander design exerts a control over the propulsion subsystem configuration in that certain emplacements are potentially incompatible with extreme ground roughness and with other Capsule subsystems. In addition, the lower surface area of the Capsule is utilized for the landing radar antenna, heat rejection devices, scientific sensing locations, and landing subsystem attachment. Engine locations must not impair these functions or overly restrict emplacements to non-optimum situations.

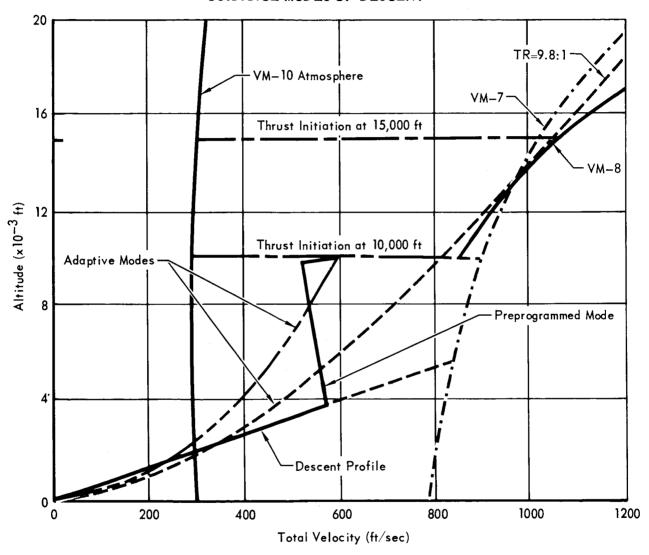
Plume impingement in normal descent, in a fire-in-the-hole separation mode, or in fire-through-holes application must not degrade Capsule Bus, Surface Laboratory, or Entry Science Package component reliability. Hence, engine size, placement and shut-off altitude all affect the decision as to which engine can be most effective for the VOYAGER Capsule Bus.

Other design requirements include:

- a. Manifolding or plumbing that minimizes center-of-gravity shifts during engine operation.
- b. Plumes must not degrade landing radar performance.
- c. Redundancies based on reliability-weight trade-offs.
- d. Adaptability to new data inputs from the Mariner 1969 flyby mission.
- e. Minimizing the potential of contaminating Martian surface while maintaining high reliability and system performance.
- f. The terminal propulsion subsystem must be compatible with the separation design.
- 4.3.3 <u>Design Approaches & Significant Characteristics</u> The method of guidance employed to meet the control requirements of the Capsule is based on surface sensing and must overcome the effects of the extremes of atmospheric characteristics. Before entering the terminal deceleration phase the vehicle's velocity will have been decreased from the entry values of 13,000 to 15,000 ft/sec down to 1500 ft/sec or less by aerodynamic drag on the Aeroshell.

Preprogrammed and adaptive guidance methods are candidates for use on the Capsule. The trajectories developed by each are shown in Figure 4.3-2. The latter requires computation of the most suitable thrust/weight value based on the sensed altitude, range, and velocity or a combination of two of these. The other, a

COMPARISON OF PREPROGRAMMED AND ADAPTIVE GUIDANCE MODES OF DESCENT



variation of a method used on the Surveyor spacecraft, has a preprogrammed set of descent maneuvers which accomplish the functional requirements of the landing approach. The sequence in the landing approach is similar in all atmospheres; however, the different atmospheres inpose a wide set of requirements on guidance sensing and upon the vehicle's controls.

4.3.3.1 Landing Approach Guidance - Terminal thrust is initiated by sensing a preset altitude. Thrust initiation may be accompanied by opening of ports in the Aeroshell to permit decelerating the full Flight Capsule, by firing rockets against the back of the Aeroshell, or by firing after separation, totally independent of the Aeroshell separation. A requirement was that the selected terminal propulsion subsystem be capable of functioning with any separation system. The first alternative necessitates a high initial thrust, the second may employ the highest initial thrust level of the engine to cause separation, whereas the third infers a low thrust level at initiation. Thus, the engines may be initiated at either high or low thrust level.

A portion of this descent will be traversed with the attitude inertially fixed or in response to a pitch-down gyro torquing maneuver. Attitude hold lasts until the Aeroshell and the lander are far enough apart axially and laterally to preclude landing radar interference. A pitchdown maneuver results in a rapid attainment of vertical orientation of the roll axis. The technique is usable only if roll control is held throughout vehicle descent.

When the landing radar detects and locks onto a reliable signal, control of the vehicle will switch from attitude hold to landing radar control, which senses range along the vehicle's roll axis and along at least three, orthogonal, doppler velocity components. The lateral velocity component is eventually canceled even if winds change the direction of the vector. The resulting gravity turn accomplishes the alignment of the vehicle to the vertical. Vehicle verticality is second in importance only to deceleration to the termination velocity.

The intermediate portion of descent can be traversed at high or low thrust depending on the form of guidance used. In an adaptive mode a self-computed thrust-to-weight trajectory would be followed; in a preprogrammed mode a preselected, low thrust level (a = 0.8 $\rm g_M$) would be employed down to a switching line. The engine selected must be capable of accommodating both types of approach modes for flexibility of mission operation. The final powered portion of the descent is accomplished at a constant nominal velocity of 5 ft/sec from 50 ft down to the 10 ft altitude

cut-off condition. The attitude of Capsule is held fixed by gyros. Because of signal noise close to the surface, the radar errors may be too great for adequate control at low velocity. A typical value of signal-to-noise ratio for the LM radar unlock is 4 dB; this value will probably be met or exceeded at rock strewn sites. Thrust is terminated by a range signal or by a probe. The free fall period minimizes engine contamination of the surface.

This portion of the Capsule Lander descent will be used in any approach mode as it fixes the conditions for the final drop to landing. Landing on the defined 34° or 0° ground slopes results in variations in touchdown of $V_{\rm v}=20$ and 16 ft/sec and $V_{\rm h}=5$ and 10 ft/sec, respectively. (See Section 2.3.7 for all values). 4.3.3.3 Alternative Engine Configuration — Several alternative configurations can meet the functional and technical requirements of the Capsule Bus and its alternative methods of separation and guidance. Of particular interest are these configurations:

Number of Engines	Additional Equipment Alternatives
One Liquid Propellant	Jet Vane Thrust Vector Control Separate reaction control jets
Three Liquid Propellant	One engine rotatable (gimbaled)
Four Liquid Propellant	Engines alternately canted
Six Liquid Propellant	One ring with all 6 required (canted or gimbaled) One ring with 5 required (2 engines gimbaled)
	Two rings; one on standby (redundant ring)
	Two rings; both active but one to be shutdown if failure occurs in final phase of terminal descent
One Solid Plus 6 Mono- propellant Verniers	Vernier engines in pairs but one ring

These variations, when combined with three modes of approach to landing and two guidance techniques, resulted in an unmanageable number of variations; so, only 18 configurations were evaluated. These are summarized in Figure 4.3-3. Four of these were selected as the high value techniques and were treated in depth. Some characteristics of each major configuration are catalogued in succeeding sections for clarification of its capabilities and limitations. The pressurant and propellant tanks were treated as independent of influence on this study.

							 _	
1	NO. OF	PROPELLANT	ATTITUDE		PPROACH MO	DE	GUIDANCI	E MODE
NO.	ENGINES	TYPE	CONTROL	SEP- ^(d) ARATION	DE- ^(e) LAYED	RE- ^(f) TENTION	PREPRO- (g) GRAMMED	ADAP TIVE
1	1	Bipropellant Liquid	Jet Vanes	х			х	
2	1			x				х
3	1					Х	0	х
4	1 (+8)	Bipropellant + (Bipropellant)	Reaction ^(a) Control System	х			х	
5	1 (+8)			х			х	
6	1 (+8)					Х		х
7	3	Bipropellant	Inherent (b)	х			х	
8	3				x		х	
9	3					Х	х	
10	3					Х		х
11	4		Inherent (k)	Х			х	
12	6		Inherent (c)	х			х	
13	6			х			х	
14	6				х			X
15	6			х			х	
16	6					Х		X
17	1 (+6)	Solid (Mono- propellant)	Verniers (c)	Х			х	
18	1 (+6)					X.	х	

(a) 8 jets — tankage common

(b) Differential thrust (P, Y); swiveled engine (R)(c) Differential thrust (P, Y); swiveled pair of engines (R)

(d) Assumed at 10 k ft and 600 ft/sec

(e) Delayed to 50 ft and 5 ft/sec

(f) Retained aeroshell to touchdown

(g) Hi-lo-hi deceleration

(h) Constant deceleration per computation

(i) A

(i) S

(k) [

(1) \$

Figure 4.3-3

4-84-1

	REDUN	DANCY USAGE	THRUST	IMPULSE	WEIGHT	VOLUME	RELIABILITY	DEVELO	DPMENT	
(h)	RINGS	USAGE	(lbf)	(lb-sec)	(lbw)	(ft ²)	(FAILURE RATE)	TIME (yr)	COST \$ M	RATING
	-		7000	80,500	737	12.79	.99129 (.009)	2,5	70.8	
	<u>-</u>		7000	85,000	762	13.20	.99129 (.009)	2.5	70.8	
:	<u>-</u>		9000	116,000	952	19.73	.99129 (.009)	2.5	70.8	
	-		7000 55/RCS	80,500	694	12.79	.99176 (.008)	2.5	75.9	6
	-		7000 55/RCS	85,000	719	13.20	.99176 (.008)	2.5	75.9	9
	_		9000 45/RCS	116,000	909	19.73	.99176 (.008)	2.5	75.9	
-	-		2330(x3)	80,500	705	10.85	.98 <i>4</i> 21 (.016)	3.5	71.6	5
	-		3000(x3)	116,000	935	13.86	.98421 (.016)	3.5	73.8	
	_		3000(x3)	110,000	905	15.58	.98421 (.016)	3.5	73.8	
	_		3000(x3)	116,000	935	16.10	.98421 (.016)	3.5	73.8	
			1650(x4)	111,000 ⁽¹⁾	-785	11.83	.97994 (.020)	3.5	65.8	2
	1		1160(×6)	96,600	843	12.43	.99720 (.003)	3.5	61.7	1
	2	Active (i)	1160(×6)	96,600	843	12.43	.99764 (.002)	3.5	61.7	3
	1		1500(×6)	139,200	1138	17.53	.99720 (.003)	3.5	64.9	8
	2	Standby ^(j)	2330(×6)	80,500	936	13.81	.99792 (.002)	3.5	71.6	4
	2		3000(×6)	116,000	1224	22.3	.99742 (.002)	3.5	73.8	10
				102,500	720	16.2	.99497 (.005)	3.0	49.8	7
			7440(s) 465(x6)	139,900	955	22.4	.99497 (.005)	3.0	50.6	

VII on; shutdown one ring in constant velocity phase only witch from one-to-another ring

Differential thrust (P, Y); alternately canted differential

hrust (R)

jized for larger vehicle; special case

(m) Propellant margin is provided for the case where one engine valve is failed closed.

Selected for further study.

<u>Single Liquid Propellant Rocket Engine</u> - Two single engines now in development are considered to be modifiable for Capsule Bus terminal propulsion. These are the Lunar Module Descent Engine (LMDE) and the Lance Sustainer.

A single engine can be gimbaled fore or aft to produce pitch or yaw control moments. If so designed, roll control can be added by cold or hot gas jets. If not gimbaled, it can produce pitch, yaw, and roll control torques by using jet vanes mounted in the nozzle or by employing a separate eight-jet bipropellant reaction control subsystem that is either an integral part of the engine's propellant supply or separate from it. Because it is the more completely developed engine, the single engine assumed in all cases is the LMDE modified to include appropriate thrust vector control. One type is jet vanes; the other is an integral 8-jet RCS. Three Liquid Propellant Rocket Engines - Three engines are needed to control pitch, and yaw and if one engine is gimbaled, roll. The engines, each capable of operating over a 10:1 throttling range, are located on or near the periphery of the Capsule Bus. Pitch and yaw attitude is controlled during descent by differentially throttling the three engines. Thrust level control is accomplished by variable area injector coupled, perhaps, with a cavitating venturi as in the LMDE. since this and all subsequent multiple engines are new developments, the choice of throttling technique is open to subsequent engineering analysis and refinement. One engine is pivoted normal to a radius line to facilitate roll control. discussion of techniques are presented in Section 5.13.3.

Four Liquid Propellant Rocket Engines - The four engine configuration differs from the three engine in individual sizes and in the form of roll control. The engines are smaller in thrust level by the inverse ratio of the number. Roll control is obtained by canting the thrust chambers in the tangent planes by 2° to 10°. Four engines provide no engine-out redundancy but obviate the need for a gimbal and gimbal actuator. A cant angle of 10°, if required, results in a constant 1.5% loss in thrust parallel to the roll axis.

<u>Six Liquid Propellant Rocket Engines</u> - The six engine configuration can provide engine-out redundancy. Several forms of redundancy were considered. Two rings of three engines per ring can be operated with one ring firing and one ring in reserve or standby; the engines in both rings are therefore equal in thrust and size to the three engine configuration discussed before. Another tehcnique consists of having all engines on but permitting selective shutdown of a ring during the final constant velocity phase where high thrust is not a requirement; failure sensing or different

dead bands are required.

Operation of all six on one ring facilitates one-engine-out capability without the weight penalty of designing to 3-engine size. With all engines operating, but sized for operation as five engines, shutdown of any failed engine simultaneously with a shutdown of the opposite engine provides redundancy throughout the mission. A detection and logic is required to sense a failure (high or low thrust) to make an appropriate gain change in the control logic.

One Solid Propellant Plus Six Monopropellant Engines - A single, spherical solid propellant rocket for an initial high thrust level deceleration coupled with six, monopropellant, vernier rockets provides a high initial thrust level deceleration with attitude controlled by the verniers. Subsequent to the termination of the solid propellant rocket's thrust, the vernier rockets will decrease total velocity down to the constant velocity descent phase.

4.3.4 Evaluation - The evaluation proceeded in two steps. First, the 18 configurations were reduced to 4 high value candidates. In Figure 4.3-3 the configurations are described and the inputs for selection are tabulated. The selection factors used in the selection are listed in Figure 4.3-4 along with the weighting factors for each. Finally, the relative ratings and the numerical results of the ranking are shown in Figure 4.3-5. As seen, four candidates were selected for in-depth study. A fifth, the single solid plus verniers, was also selected for some additional consideration as a proplusion only concept of some merit in a subsequent trade study (Section 4.5).

In the second step, the selection factors were recodified under the five selection criteria, and each engine configuration's capability of meeting these criteria was assessed.

4.3.4.1 <u>Mission Success Evaluation</u> - Probability of mission success is enhanced by the use of highly reliable, simple circuits in subsystems, careful failure mode and effect analysis with design feedback, judicial employment of redundancy, and designs aimed at performing in the worst of anticipated environments. The last item is implemented by operating the engine in a derated condition if the nominal environment is encountered. Probability of mission success is a function of deleterious actions by the propulsion system on other portions of the Capsule Bus or to the Martian surface in close proximity to the landing site.

The reliability analysis consisted of apportioning unreliability by judgement between all of the Capsule Bus subsystems and, then, examining those that have a

SELECTION FACTORS AND WEIGHTING FACTORS

Probability of — Attenuation of signal, phase shift, and bending of beam

Mission Success - Relative power backscatter

(0.35)

- Damage and heating at separation

- Additional development because of configuration

Mars surface impingement
Effect of rocket arrangements
Probability of landing success
Reliability of components

- Complexity of subsystem or of the entire system

System Performance

(0.20)

- Weight of complete system

- Size of components

Effect on over-all vehicle weightEfficiency of engine operation

Effect of initial and terminal conditions
Extreme use of lower surface area

- Length of engines

Lander design compatibility
Obstruction of the center line
Location of lander in the aeroshell

- Requirement for porting (retained and delayed)

Development

Risk (0.20) Confidence in meeting required schedule
 Difficulty of proof testing and simulations
 Experience with other state-of-art projects
 Scheduled time to complete development

Versatility (0.15)

- Versatility in dealing with atmosphere extremes

Response to undefined operational inputs requirements
Capability of accepting late requirement changes
Adjustability in the late stages of operations

- Ability to accept larger payloads

- Potential for corrective actions during mission

Costs (0.10)

- Cost estimates through qualification testing

- Hardware procurement (12 sets)

Ratings: .1 to 1.0 with 1.0 high

 $Maximum \sum f_i = 1.00$

S - Separated R - Retained

Designations:

D – Delayed P – Programmed A – Adaptive

1 – One Ring 2 – Two Rings

* — Active ** — Standby

RATINGS			-	9	6		<u> </u>					8	က		4	01		
TOTAL	.42	4.	.35	09:	.54	.44	.61	.45	.45	.50	69.	.70	89:	35.	.65	.52	.59	05.
VERSATILITY (.15)	60. 9.	60. 9.	.5 .08	.7	60.	.5 .08	.8 .12	60. 9.	60. 9.	.5	11. 7.	.8	9.	.7.	.9 .14	60. 9.	.7.	60. 9.
COST (.10)	90. 9.	90. 9.	90. 9.	90. 9.	90. 9.	90. 9.	.5 .05	4.04	.4 .04	.4 .04	70. 7.	90. 9.	.4 .04	90. 9.	.4 .04	.4 .04	70. 7.	70. 7.
RISK (.20)	.2 .04	.4	.3 .06	.4 .08	.3 .06	.4 .08	.6 .12	.3 .06	.3 .06	.4 .08	.4 .08	.4	.4	.4	.4 .08	.4 .08	.4 .08	3 .06
PERFORMANCE (.20)	.12	.12	40.	4.	.12	.04	81.	.12	.12	.12	81.	91.	.14	9:	1.	٥.	.12	01.
PERFORM	9.	9.	.2	.7	9.	.2	6:	9.	9.	9.	6.	œ	۲.	.5	7:	٠.	9	ئ.
SUCCESS (.35)	Ξ.	=	Ε.	.21	.21	. 18	81.	1.	4.	. 18	.25	.28	.78	.21	.25	.21	.21	. 18
SUCC	ε.	ь.	ω	.	9	٠.	5.	4.	4	z.	7:	œί	7.	٠.	7.	6	9.	5.
USAGE	I	í	l	ı	ı	ı	ı	ı	ı	ı	ı	1	*	ı	*	*	ı	ı
RINGS	ı	i	ı	1	ı	ı	1	1	ı	ı	ı	-	2	_	7	2	ı	ı
G&C	۵	∢	∢	۵	۵.	∢	۵	α.	<u>a</u>	∢	۵	۵.	<u>a</u>	∢	<u>_</u>	∢	۵	Д
APPROACH	S	S	œ	S	S	œ	S	۵	œ	œ	S	S	S	۵	S	C .	S	~
ENGINES		_	_	80 + -	1 + 8	1 + 8	ო	ო	ო	က	4	9	9	9	9	9	1+6	1 + 6
NO.	_	7	က	4	2	9	7	œ	6	2	=	12	13	4	15	91	17	18

Figure 4.3-5

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direct influence on the selection of an engine configuration. Thus, the effects of guidance and control sensors on the engine and vice versa were included. In like manner, the mode of landing approach and the mode of guidance were considered for their effect upon reliability. When the effects of radar and the inertial sensors were assessed, only a weighted percentage (based on the unreliability apportionment) was included.

The cutoff point on depth of reliability analysis was a failure mode, effect and criticality analysis performed on the four, high value propulsion configurations discussed above. In fact, only those failures that are considered catastrophic in nature have an influence on the selection. All other failures are performance degradations which may be eliminated as the engine is developed. Most catastrophic or single point failures can be circumvented; however, the more there are, the lower the confidence level of the computed reliability value.

The findings relative to mission success are:

- a. Canted engines eliminate the complexity of gimbals and actuators, so 4 and 6 engines have a reliability gain.
- b. Fire-through-holes with exhaust ducts and with retention of the Aeroshell to low altitude is less reliable than other separation techniques. A single engine is more reliable than are multiple engines.
- c. System reliability is highest for a six engine configuration with only five engines required but the required detection and logic equipment reduces its estimated reliability to a level comparable with four.
- d. Reliability improvement is markedly improved if a suspected non-operating failure mode is eliminated (degradation due to vacuum exposure of ablative, throttleable engines). The 6-engine (5 required) has the highest reliability but the reliability-weight relationship (improvement/pound) is superior for all other engine configurations.
- e. Contamination levels and erosion altitudes (site alteration potentials) are highest for a 1-engine configuration and decrease inversely with engine numbers.
- f. The landing radar antennas are least affected by a 1-engine configuration; however, the double antennas are less reliable than is a single phased array. ($\Delta R = -.0012$).
- g. The inertial sensors can be located optimally at the center of mass in 3,4, or 6 peripherally mounted engine configurations; non-optimally in a1-engine or multi-engined, clustered configuration.

The order of overall perference based on mission success is - 1, 4, 3, 6 - unless a simple logic can be devised for sensing an engine failure and its mode for a 6-engine (5-required) configuration. If devised, the 6-engine configuration jumps to first.

4.3.4.2 System Performance Evaluation - Performance of the Capsule Bus decelerators is gaged by how large an increment of the entry mass has to be devoted to the incremental deceleration. Another indicator is the Capsule's controllability. Performance of the terminal propulsion is measured by its weight, by the volume it occupies, by the constraint or restriction it places on equipment locations within the vehicle, and by rapidity of achieving Capsule verticality. To limit the selection to differences in the number and form of engines, all components were standardized wherever practicable in this study. The summed values of propellants, tank dry weights and engine weights are indicators of performance, as is the volume required to package the subsystem.

Comparisons between the different methods of approach, guidance modes, thrust initiation altitudes, and engine configurations were included to select between all influencing factors. Total impulse values, as listed in Figure 4.3-6, were originally selected near the point of minimum impulse required in the case of a separated lander. Subsequent analyses showed that a higher separation altitude was needed to permit complete clearance of the radar beam from in front of the Aeroshell for fire-in-the-hole separation. Figure 4.3-7 illustrates a possible installation of the single engine in a Capsule Lander.

The single engine installation, as compared to the subsequently selected baseline 4-engine configuration, has the following disadvantages:

- a. 5% heavier foot pad and platform due to hole in center.
- b. Increased Surface Laboratory (cold) insulation required.
- c. Non-optimum gimbal usage (led to preference for jet vanes or RCS).
- d. Crushable nozzle extension required.
- e. Raised center of gravity degrading stability.
- f. Incompatible with preferred rover configurations.
- g. Parachute forced off center line; there is no other good position since a mortar must be employed.
- A single engine has these advantages:
- a. Facilitates propellant mass balance.
- b. Radiant surfaces of Surface Laboratory are unobstructed.
- c. Split SLS facilitates surface sampling over the edge of the Capsule Lander platform.

SUBSYSTEM PERFORMANCE

NUMBER OF ENGINES	THRUST (LBF)	IMPULSE (LBF-SEC)	WEIGHT (LBW)	VOLUME (FT ³)	DENSITY (LBW/FT ³)	TYPE
1	7000	110,000	844	15.3	55.1	(S)
3	2330	110,000	863	12.8	66.8	(S)
4	1650	110,000	875	11.8	74.1	(S)
6	1160	110,000	921	13.6	67.0	(S)
1	9000	170,000	1187	24.3	49.0	(R_A)
3	3000	150,500	1119	19.0	58.7	(R)
6	1500	170,000	1291	20.1	64.2	(S_D)

(S) SEPARATION: 800 FT/SEC OR 15,000 FT ALTITUDE | Isp = 288 SEC

(R) RETENTION: TO SURFACE

(-D) DELAYED: 5 FT/SEC OR 50 FT ALTITUDE

(~A) ADAPTIVE CONTROL

SINGLE ENGINE LANDER LAYOUT

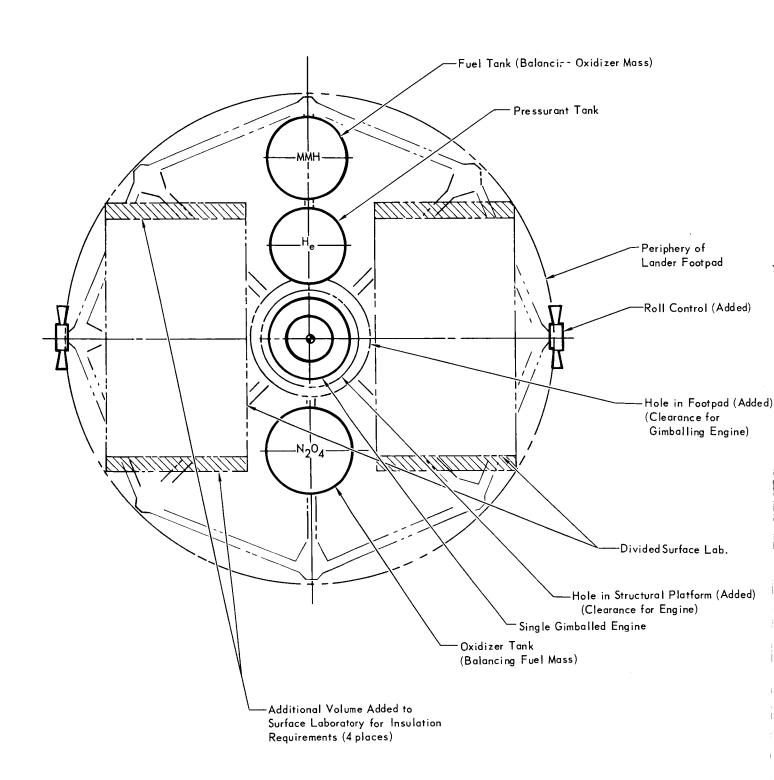
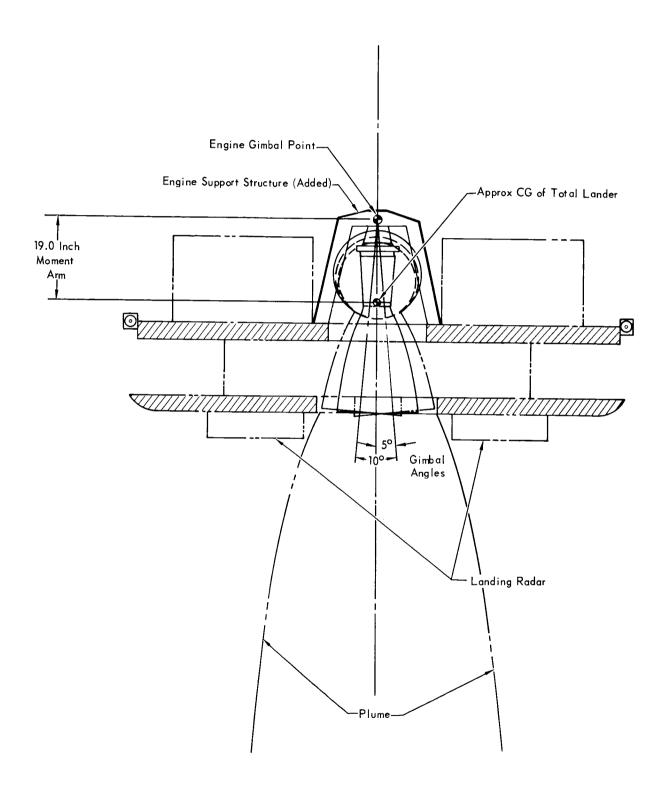


Figure 4.3-7

4-92 -/



The conclusions drawn relative to system performance are:

- a. The single engine is lightest but the four engine configuration is light and requires the least volume.
- b. The three engine occupies the least amount of bottom surface area with the four engine configuration slightly larger.
- c. The six engine configuration has the shortest engines and the least diameter engines but has the maximum exit area.
- d. Of the multiple engine configurations the three engine configuration is lightest because it provides roll thrust on demand only. Thus, it has no propellant weight penalty associated with it, whereas the canted engine roll control will have up to a 1.5% loss even if roll commands are absent.

Recommended in order: 4, 3, 6, 1.

4.3.4.3 <u>Development Risk</u> - Development risk is assessed by determining the state of development of existing engines or lacking applicable engines, by evaluating the state of the art of sterilized throttling engines. The two designs that may be used as a basis for the VOYAGER Capsule Bus are the Lunar Module Descent Engine (TR = 10:1) and the Lance missile sustainer engine (TR = 50:1). Each require extensive modification in such characteristics as:

- a. Pressure level to adapt to Martian atmospheric pressure conditions.
- b. (Resizing of the thrust chambers because of chamber pressure and exit pressure changes).
- c. Upstream flow passage modifications to account for pressure mixture ratio, flow rate changes, and propellant changes.
- d. Sterilization requirements.

Although not a part of the engine development, a more involved radar antenna development problem accompanies the use of a single engine. Split antennas were used in the Surveyor lunar landing radar but the performance of those antennas is considered inferior to that of the phased array used in the Lunar Module. Since all antennas are being developed, the difference probably would be negligible. Installation and alignment of a split antenna is more difficult but considered to be straightforward, hence, not a development risk.

Another form or development risk is complexity. A single engine with expansion cone liquid injection or jet vane thrust vector control is least complex and, hence, a low risk. A single engine with an integral eight-jet reaction control subsystem is more complex than three and four-engine configurations. The

potential advantage of one-engine-out capability latent in the six-engine configuration tends to be by the requirement for failure sensing and for extensive simulation testing. A diluted network that can sense an excessive differential (high or low) from the nominal and can detect engines that are shut-down or wide open at a given time are complex and doubtful of solution within the development time span of three and one-half years that would have to be imposed.

Summarizing the development risk evaluation:

- a. The single engine (without TVC) is the lowest development risk. The four-engine configuration is probably a lower development risk than the three-engine configuration because gimbaling is not required.
- b. A high altitude separation provices the maximum time for maneuver and radar lock-on; hence, it has less critical timing, which results in low development risk. Delayed separation is next because separation is at low speeds (hovering). The retained mode is the most difficult to simulate and the landing and leveling requirements are extreme, so the retained mode has the greatest risk. Thus, fire-through-hole implementations were de-emphasized in the final selection of configurations.

The order of preference is 4, 3, 1, 6-engine configurations with preprogrammed descent from a moderately high altitude (5,000-15,000 ft).

4.3.4.4 <u>Versatility</u> - Versatility, as applied to the terminal propulsion subsystem, means adaptability to new requirements late in the development stage caused by changes in atmospheric definitions. An allied requirement is growth potential. Growth refers to the ability to grow from a weight at thrust initiation in 1973 of 2650 lbs to a weight of 4170 lbs in 1979. The 57% increase in weight must be accomplished with essentially the same engines in 1979 that will be used in 1973 if the requirements of maximized standardization are to be met.

Uncertainty in atmosphere and surface conditions can be accommodated by designing into the terminal propulsion subsystem some margins of performance. Actual changes in thrust or response may be required from opportunity to opportunity. Judicious balancing of capabilities and requirements must be practiced.

Two techniques for growth are readily available: (1) increase thrust to maintain the thrust-to-weight ratio which is the antithesis of standardization, and (2) size the thrust chambers and the throttling ratio for the minimum thrust level expected (.8 $g_{\rm M}$ @ 2650 lbs Earth weight in 1973) and make the throttling ratio sufficiently high to assure satisfactory deceleration in the maximum weight case (4170 lbs in 1979). This entails a reduction in the thrust-to-weight ratio; but,

if properly analyzed, the growth can be accommodated.

Thus, margins of rocket engine thrust, duration, throttling range, and other operational adaptability will be judiciously incorporated. An example of an engine already possessing a thrust margin - in fact, two margins - is the LMDE. One margin is a 9700 lb constant thrust level; the other is a capability to be operated from 6300 down to 1050 lbs thrust continuously. This appears to be an adequate thrust and throttling range for both the 1973 and 1979 missions.

The six-engine configuration can also be operated in steps. In the programmed descent phase all six engines can be fired simultaneously. Before and after this variable thrust portion three engines could be shut down, providing an additional two-to-one step capability. This procedure is not recommended for the Capsule Bus as 10:1 throttling appears to be sufficient and more flexible and decreases the likelihood of failure in switching engines on or off.

Thus, all engines considered have inherent or designed-in capability beyond currently predicted maximum requirements; however, the six-engine configuration is most adaptable to new data inputs because of inherent versatility. Based on this, the order of preference is 6, 4, 3, 1.

4.3.4.5 Costs - The costs of developing and procuring the twelve sets of engines that are needed for the 1973 opportunity are comparable for each of the configurations considered in the final selection. (See Figure 4.3-8). Two possible cost structures are involved in the single engine evaluation. The first is use of a modified (minimum change) Lunar Module Descent Engine (LMDE): \$81.2 million. If this is unmodifiable for any reason and the development needs completely new designs, the cost is considerably higher: \$99.7 million.

It is noteworthy that when the modified engine is combined with a reaction control subsystem (RCS) to control attitude, the cost is higher than that of newly developed, multiple engines. The increased amount is almost small enough (2% max.) to be unimportant if only the single and the average of the several multiple engine costs are considered. However, the key factor is the difference in magnitude of the terms "modified" and "redesign". In the engine under consideration, as discussed previously, there is really a redesign of the engine's exterior, of its injector periphery, and of its pressure level (300 instead of 100 psia chamber pressure). All three changes result in significant changes in performance, in development risk, and in mission success probability. They also reflect in development costs. The two-to-one ratio between engine development and modification costs may be high and subject to reduction as design requirements on multiple engines

DEVELOPMENT AND PROCUREMENT COSTS

NUMBER OF ENGINES		DERATED (<u>MILLIONS</u>)	REDESIGN (<u>MILLIONS</u>)	NEW DESIGN (<u>MILLIONS</u>)
1	Engines: (D) (H) RCS: (D) (H) Plumbing: (D) (H) Totals	16.0 3.6 	34.6 3.5 \$99.7	12.0 13.2 23.8 12.6
3	Engines: (D) (H) Plumbing: (D) (H) Totals			48.0 5.7 18.3 7.4 \$79.4
4	Engines: (D) (H) Plumbing: (D) (H) Totals			46.8 6.8 19.1 7.8 \$80.5
6	Engines: (D) (H) Plumbing& (D) detection: (H) Totals			42.0 8.4 20.9 8.5 \$79.8

(D) Development: Design thru qualification (H) Hardware: 12 sets

are firmed. A difference in favor of multiple engine configurations will persist, however.

Thus, each of the multiple engines is less expensive than is an average value of a modified/redesigned single engine configuration. Another cost factor to be considered, though of relatively low magnitude compared to \$80 million, is the fact that a more difficult, and hence more expensive, landing radar antenna development would be incurred with a single engine configuration.

The six-engine configuration that was priced is the one requiring only 5 engines operating. The detecting and logic costs are included. Even so, the six-engine configuration is not the most expensive. Again, one-and-a-quarter percent spread from least to most expensive is not sufficient to affect a decision for or against any one of the three multiple engine configurations if strong reasons exist for a particular design. On the basis of estimated costs, the preference is 3, 6, 4, 1.

4.3.5 Recommendation - Combining the ratings from each selection criterion in accordance with the weighting values shown in Figure 4.3-9 leads to the recommendation that the four-engine configuration be selected. The most important single factor in this selection is the compatibility of this system with the lander configuration. Had the single engine configuration been equally compatible, it would have rated as high overall as the four-engine configuration and probably have been selected on the basis of its slightly better rating for mission success.

VERSATILITY	Most adaptable to Lander configuration. Can be modified in thrust level with little effect on Capsule design. 10:1 throttling adequate.	Adaptable to Lander configurations. Low adaptability of roll control owing to engine gimballing. 10:1 throttling provides adequate growth.	Least adaptable to preferred Lander configuration. Control location makes redesign of Capsule the most involved. 10:1 throttling provides adequate growth capability.	• Step operation capability provides added adaptability. • Most adaptable to 1969 and 1971 data inputs. • Easiest to modify thrust level. • Step operation could be used to reduce 10:1 continuous throttling, or to improve growth potential.
COST	Low engine cost because gimbals not needed. Highest cost multiple engine system (\$80.5 million) (Difference may be negligible or may reverse order).	Least cost system; at low point between engine costs and system costs (\$79.4 million). High engine unit costs.	Requires development of two engines – main and RCS. Potentially the highest cost (\$81.2-\$99.7 million). Requires split landing radar antenna development.	• Requires development of a failure detection and logic net to use 5 out of 6 engine capability. • Engine (only) development costs least because they are smallest. • Low costs (\$79.8 million). • Gimbal costs.
DEVELOPMENT RISK	Simplest system so lowest risk. No gimbal required. Predicted development time (3.5 years). Least checkout time in system test and pre-launch.	Simple system so low risk. Requires development of engine gimbal. Predicted development time (3.5 years).	Requires auxiliary attitude control capability or roll control and TVC. Requires modification of pressure level, all material, flow path changes. Check of sterilizability. Predicted development time (2.5 years). Requires installation of split antenna.	Most complex system especially if one-engine-out capability (failure detection). Gimbal required. Predicted development time (3.5 years). System tests most complex and facility requirements most severe.
SYSTEM PERFORMANCE	• Heavier (875 lb). • Least volume (11.8 ft³). • Small exit area (1.97 ft²). • Small combined area required (5.27 ft²). • Short engine (25 in.). • Small diameter (9.5 in.)	• Slightly heavier (863 lb). • Slightly larger (12.8 ft ³). • Small exit area (1.87 ft ²). • Smallest combined (5.17 ft ²) area. • Moderate length (28 in.). • Small diameter (11 in.)	• Lightest system weight (844 lb). • Largest volume (15.3 ft ³). • Minimum engine exit area (1.77 ft ²). • Maximum combined radar and exit area (5.87 ft ²). • Longest engine (38 in.). • Largest diameter (18 in.)	• Stepped throttling operation is feasible. • Heaviest (921 lb). • Large volume (13.6 ft?). • Largest exit area (2.64 ft?). • Largest combined area required (3.94 ft?). • Shortest engines (19 in.). • Smallest diameter (9 in.).
PROBABILITY OF MISSION SUCCESS	• More engines; less effect from misalignments. • Roll control by canting engines is possible (20-10°). • Requires tight tolerance on ignition delay. • Most compatible with landing radar (3 or 4 beam). • Best engine size/system complexity.	Multiple engine misalignments have less severe effect. Simplest. Requires tight tolerance on ignition delay.	• RCS has one engine out reliability. • Reduced reliability of the radar because of split antenna. (AR = .0012). • Least tip-over at ignition. • Longest plume – affects aeroshell longer and ground sooner. • Maximum component reliability. • Maximum contamination potential • Least effects on radar. • Forces inertial	Most engines; least effect from misalignments. Provides engine-out capability; needs a failure detection and logic net. Most complex Requires tight tolerance on ignition delay. Minimal contamination potential.
	Four LPR Engines Configuration	Three LPR Engines Configuration	One LPR Engine Configuration Selected Configuration	Six LPR Engines Configuration

4.4 DEPLOYABLE AERODYNAMIC DECELERATOR - An auxiliary aerodynamic drag device is desirable to decelerate and stabilize the Capsule to provide more suitable conditions for the terminal propulsion subsystem (Section 4.5). The drag force produced by the decelerator is also used to separate the lander from the Aeroshell (Section 4.6).

The relative merits of two auxiliary aerodynamic decelerator concepts - parachutes and Ballutes - for the 1973 VOYAGER mission were investigated. After analysis of both concepts, the supersonic solid parachute was selected.

4.4.1 Operational Requirements - The mission envelope is defined by a variety of entry conditions and a wide range of postulated Martian atmospheres and wind profiles. Even though all of the postulated atmospheric models have extremely low density compared to that on Earth, 93 to 98% of the entry velocity is removed with only Aeroshell drag by the time the aerodynamic decelerator deployment altitude has been reached. However, even with an additional aerodynamic decelerator of reasonable size, impact velocities of 100 to 300 ft/sec would be expected. For this reason, and to remove ground drift caused by winds, a terminal propulsion system is required. The inclusion of a propulsion subsystem does not negate environmental conditions as a constraint since propulsion capability is limited by other factors such as weight, volume, and time available for the terminal descent maneuver.

The magnitude of the winds at low altitudes vary extensively with atmospheric model. Wind shears and wind gusts in combination with the constant wind profiles magnify stability deficiencies, tend to complicate the lander's descent trajectories, and prohibit the attainment of low terminal descent velocities which would relieve terminal propulsion requirements. Low terminal speeds are prohibited by radar considerations since, with vertical velocities lower than wind velocities, the lander must rotate to such a shallow attitude in order to fire its propulsion unit in the direction of the relative wind that radar lock with the ground would be broken. As the ballistic parameter (m/C_DA) increases, more severe heating environments and dynamic pressures are encountered. The low scale height atmospheres produces high Mach numbers and dynamic pressures at low altitudes which force an upper limit on the deployment altitude if Mach number constraints exist for the particular decelerator concept considered.

The Capsule descent trajectories vary considerably due to the broad range of postulated Martian atmospheric density and scale height and the range of entry conditions. Rigid performance requirements are imposed on the auxiliary decelerator triggering devices studied exhibited an altitude - Mach number operational envelope

too severe to warrant further consideration (Section 5.10). The radar altimeter was selected as the primary triggering subsystem because it maintains a consistent envelope of deployment conditions.

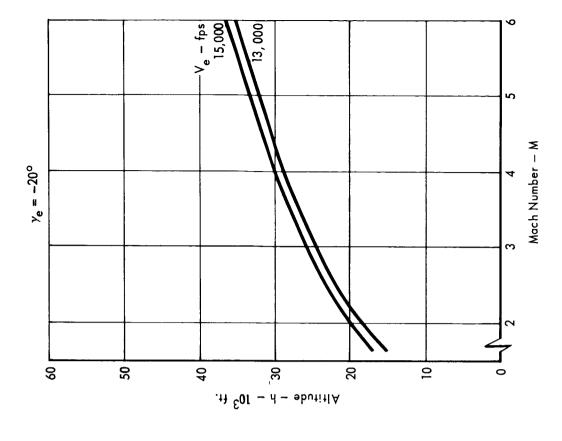
The selected decelerator must also fulfill the basic mission objectives of: (1) soft landing the Surface Laboratory System, (2) providing a stable platform for television coverage during the descent, (3) providing a satisfactory trajectory during which atmospheric data may be collected, and (4) rapid descent to allow sufficient time for post-landing data transmission. In the following paragraphs, the constraints imposed on operational performance are discussed. 4.4.1.1 Entry Corridor - The allowable range of entry velocities is 13,000 to 15,000 ft/sec at the reference entry altitude of 800,000 feet. The initial flight path angles associated with these entry velocities may range from vacuum graze to -20 degrees. The deceleration subsystem must operate in the most severe of the postulated Martian atmospheres. For a nominal m/C_DA of .3 slugs/ft² and in the VM-8 atmosphere, trajectories for an entry velocity of 13,000 ft/sec show that -20 degrees is the critical entry flight path angle (See Figure 4.4-1). flight path angle with an entry velocity of 13,000 ft/sec results in the highest Mach numbers in the altitude region of interest for decelerator deployment (See Figure 4.4-2). This trend persists with other atmospheres under similar conditions.

In Figure 4.4-3, the effect of entry conditions on dynamic pressure is presented. An entry angle of -20 degrees and an entry velocity of 15,000 ft/sec produces the maximum dynamic pressure of 202 psf. It is necessary, however, to compare the variation of dynamic pressure with altitude, shown in Figure 4.4-4, for a flight path angle of -20 degrees with entry velocities of 13,000 and 15,000 ft/sec. Again the 13,000 ft/sec velocity and the -20 degree flight path combine to produce the most severe dynamic pressure conditions at altitudes below 40,000 ft. As a result of these comparisons, the entry design conditions become 13,000 ft/sec and -20 degrees.

4.4.1.2 <u>Martian Atmospheric Models</u> - Ten Martian atmospheric models have been postulated. Surface pressures range between 5 and 20 millibars, resulting in the density variations presented in Figure 4.4-5. Two distinct families of curves are shown in the variation of Mach number and dynamic pressure, Figures 4.4-6 and 4.4-7. The upper atmospheric densities are greater in the odd numbered atmospheres, resulting in lower velocities and steeper flight paths. VM-7, VM-8, and VM-10 were used in the study, corresponding to the limiting cases shown in Figures 4.4-6 and 4.4-7; VM-9 is not a limiting case because VM-10 becomes more



EFFECT OF ENTRY CONDITIONS ON MACH NUMBER $M/C_DA = .3 \quad VM-8$



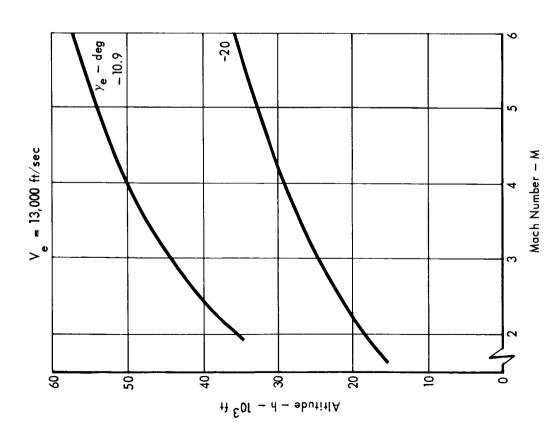


Figure 4.4-1, 4.4-2

4-101

Figure 4.4-1

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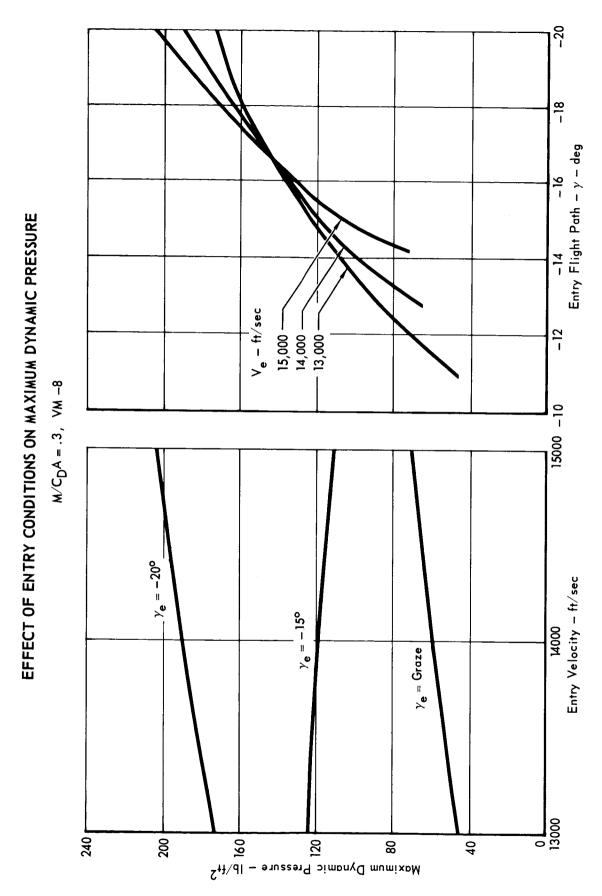


Figure 4.4-3 4-102

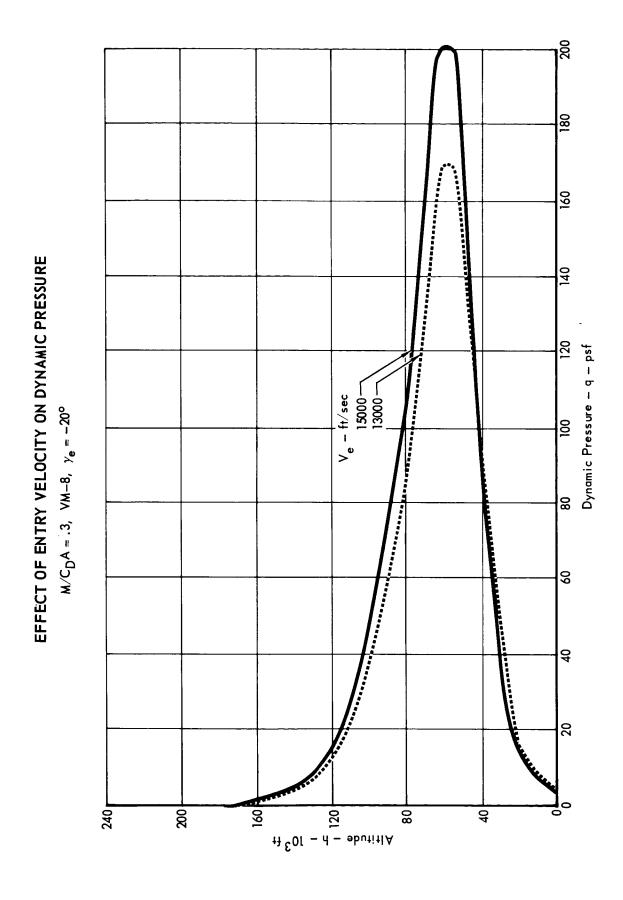


Figure 4.4-4 4-103

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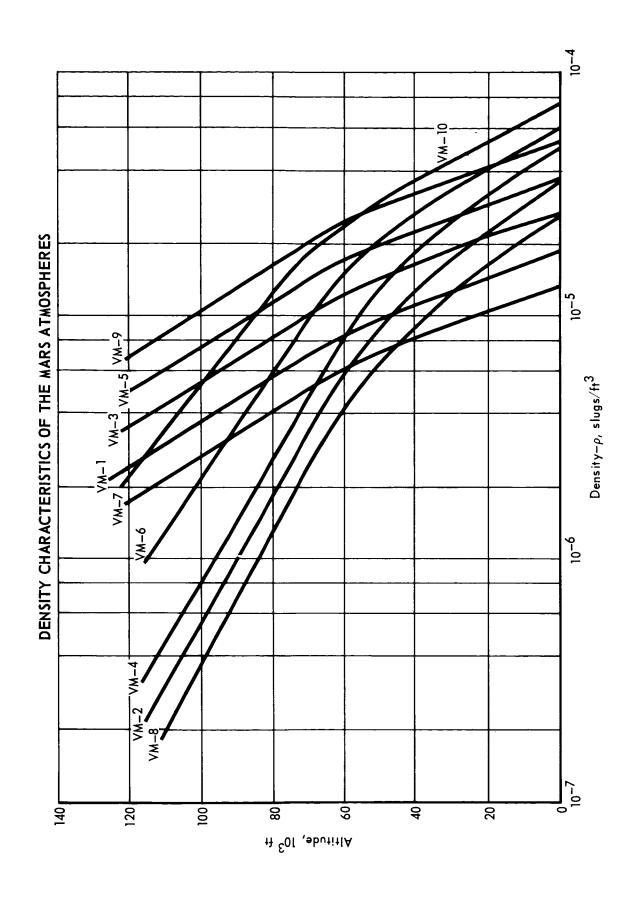
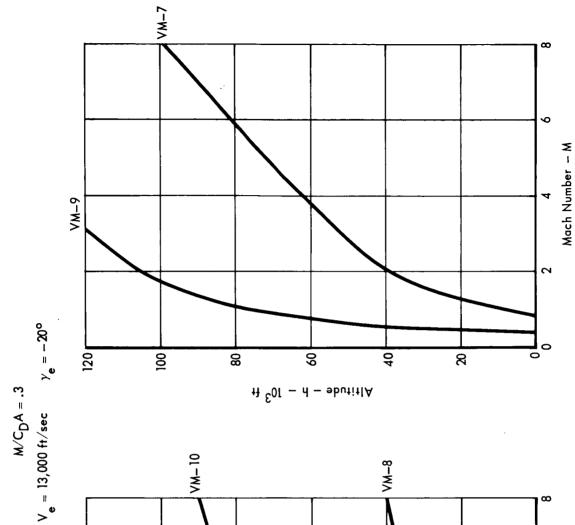


Figure 4.4-5 4-104



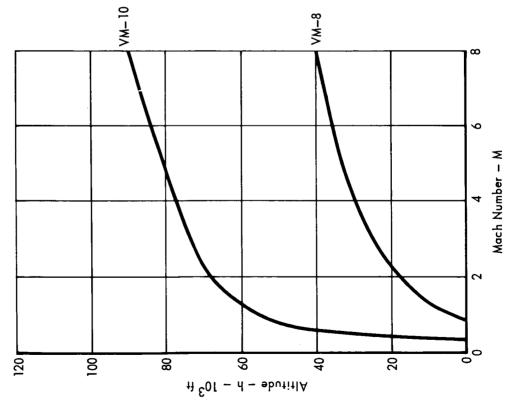
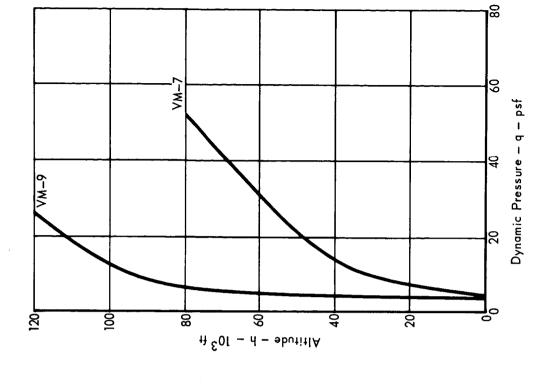


Figure 4.4-6

4-105

COMPARISON OF DYNAMIC PRESSURES IN VARIOUS ATMOSPHERES

 $M/C_DA = .3$ V_e = 13,000 ft/sec $\gamma_e = -20^o$



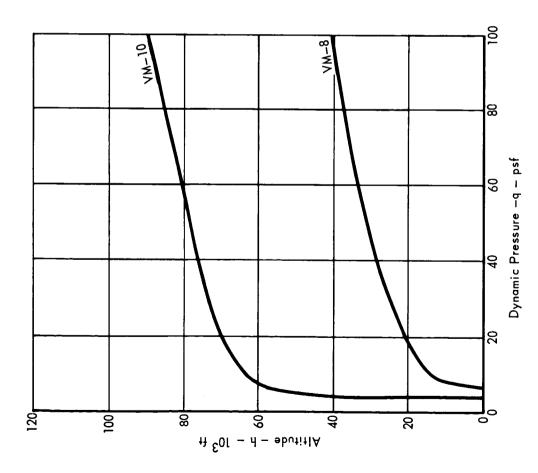


Figure 4.4_7

4-106

dense below 55,000 ft. At low altitudes VM-7 has the least density and hence higher velocity, as shown in Figure 4.4-8. (The Mach number in VM-8 is higher than in VM-7, but this is due to the variation of the speed of sound between the two atmospheres).

- 4.4.1.3 <u>Ballistic Parameter (m/CpA)</u> The mass of the Aeroshell/Lander is important in the selection of an auxiliary aerodynamic decelerator. The effect of m/C_DA on Mach number is shown in Figure 4.4-9 for an entry velocity of 13,000 ft/sec and an entry flight path angle of -20 degrees in VM-8. This figure shows that the m/C_DA could be limited by either Mach number or the minimum altitude required for proper separation of the Lander and the Aeroshell. This becomes even more important in missions subsequent to the 1975 launch opportunity when increased payloads are anticipated.
- 4.4.1.4 <u>Sterilization</u> The materials used in the construction of any deceleration device must be compatible with the sterilization requirements defined by NASA. We know of no evidence to warrant penalizing any one of the candidate decelerators because of sterilization requirements. We are confident that existing materials can be used for fabricating the decelerator.
- 4.4.2 <u>Candidate Concepts</u> Two decelerator concepts have been considered for the VOYAGER program: supersonic parachutes and Ballutes. Both ribbon and solid type supersonic parachutes, having high and low geometric porosity respectively, were investigated, along with trailing, attached, tucked-back, and Airmat cone Ballutes. The drag coefficients of the candidate decelerators are compared in Figure 4.4-10.

Trailing Ballutes and extended Aeroshell decelerators have been successfully used in several applications that required high Mach number capabilities. Ballutes are effective and reliable drag producing devices above Mach 2, and are characterized by exceptional stability. All of the parachutes considered are more effective subsonically than the Ballutes but loose their efficiency rapidly above Mach 2. Trailing Ballutes have good inflation characteristics and are well proven, reliable devices.

Although there is some uncertainty about the inflation characteristics, stability levels, opening shock loads, and operational repeatability of parachutes in the rarefied Martian atmosphere, the continuing Planetary Entry Parachute Program (PEPP) has filled some of these void areas (References 4.4-1, 4.4-2, and 4.4-3). The PEPP tests have demonstrated the feasibility of parachute operation in low densities and at supersonic Mach numbers.

One of the primary functions of the deployable aerodynamic decelerator is to

ATMOSPHERIC INFULENCE ON VELOCITY

 $M/C_DA = .266$

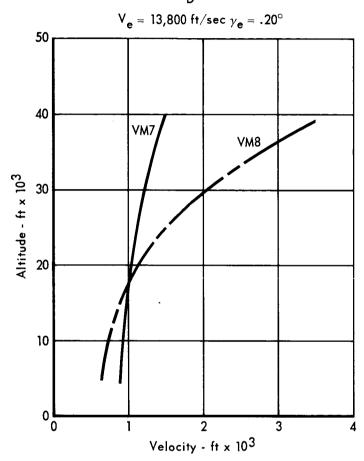


Figure 4.4_8

EFFECT OF BALLISTIC PARAMETER ON MACH NUMBER

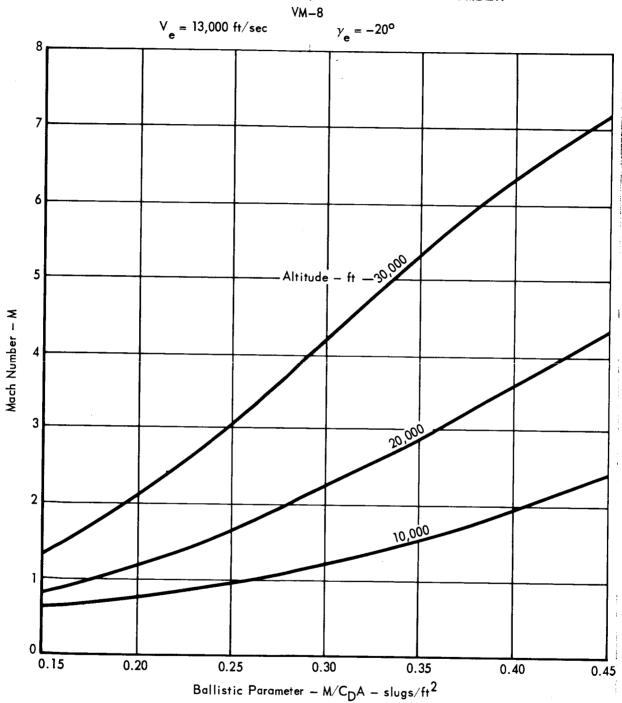


Figure 4.4-9

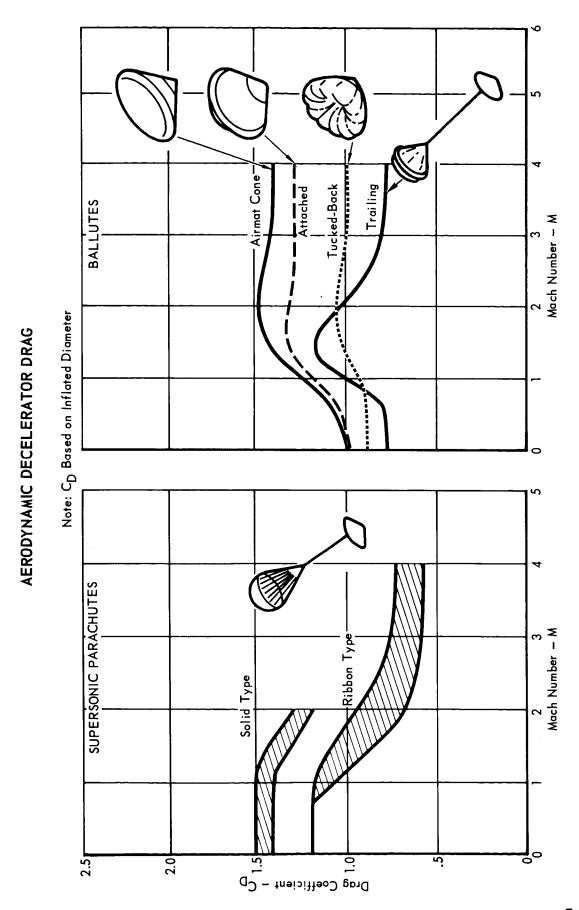


Figure 4.4-10

assist in the Aeroshell/Lander separation sequence. This mode of operation imposes requirements which, when considering the Ballute as a candidate concept, are presently of unknown consequence. A separation – recontact phenomenon has been observed in several wind tunnel tests by other agencies. The drag of the afterbody varies with separation distances as shown in Figure 4.4-11. At a certain separation distance, the drag of the afterbody decreases below the level required for continued separation and recontact occurs. This separation—reattachment motion is periodic in nature. The probability of this phenomenon occuring with a Ballute is an unknown factor which would have to be investigated during the development program.

Another unknown element related to the Ballute concerns the change that is required in attachment points, from Aeroshell to the Lander, and the lack of a solid attachment structure. When Ballute attachment to the Aeroshell is released (Point A in Figure 4.4-12), the lines from A to B are pulled taut with a resulting shock load, extracting the Lander from the Aeroshell. Without a solid attachment structure, the shape of the Ballute will change to a smaller diameter and a reduced cone angle, resulting in decreased drag. The air passageway opened along the centerline of the Ballute would then require a cover to preserve drag efficiency. These unknowns are not insurmountable problems, but simply areas where development testing is indicated.

The decelerator should provide stability to the Lander to improve the quality of data collection and television coverage during the descent as well as performing its deceleration function. Inherently, attached Ballutes are more stable — statically and dynamically — than either trailing Ballutes or parachutes, due to their more rigid characteristics. However, the trailing decelerator and Lander react to disturbances in a more flexible manner than the more rigidly connected Ballute.

Preliminary gust dynamic analyses were conducted for McDonnell by Northrop Ventura. These analyses were generalized in nature and assumed certain parachute stability characteristics which must be verified by wind tunnel tests, but the results can be considered as representative.

The dynamic response of the Lander angle of attack during the first 2 seconds after encounter with an instantaneous gust of 200 ft/sec is damped to within \pm 5 degrees of the equilibrium flight path angle. An oscillatory motion of \pm 4 or 5 degrees persists for an indefinite time period. The Lander flight path angle may be displaced initially as much as 50 or 60 degrees but then a smooth transition to

POSSIBLE TREND OF TUCKED-BACK BALLUTE DRAG COEFFICIENT WITH SEPARATION DISTANCE

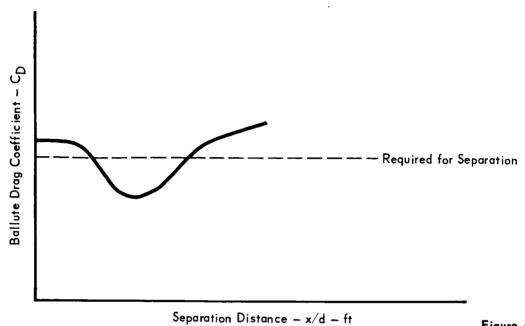


Figure 4.4—11

SECTIONAL VIEW OF TUCKED-BACK BALLUTE ATTACHMENT DETAILS

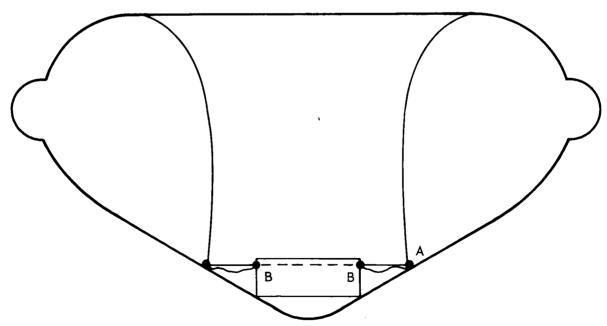


Figure 4.4-12

the equilibrium flight path occurs. The Lander attitude is initially displaced 70 or 80 degrees from vertical, but within 5 seconds is damped to a \pm 4 degree oscillation.

The angle between the parachute riser and the Lander centerline exhibits high frequency oscillations, but the amplitude seldom exceeds ± 5 degrees. This oscillatory motion, however, sometimes persists as long as 30 seconds, thus affecting the mission requirements for atmospheric measurements, television pictures, and data transmission. Various attachment locations and riser suspension network configurations must be explored to reduce the severity of the oscillations. It is emphasized that the gust was applied instantaneously in the above studies. A sharp-edged gust of such magnitude is probably unrealistic; even a 1 second onset would substantially reduce the dynamic response.

4.4.3 Evaluation - To aid us in our evaluation of the two decelerator concepts, we contracted Northrop Ventura to conduct a parachute analysis and Goodyear Aerospace Corporation to conduct a Ballute analysis. These are detailed in References 4.4-4 and 4.4-5. We have also conducted a preliminary Ballute wind tunnel test to aid in evaluating this concept. The test was conducted in the Trisonic 4 ft. wind tunnel at Douglas El Segundo in July 1967 (Reference 4.4-6). Schlieren photographs and pictures of two basic models are presented in Figure 4.4-13. The relative advantages and disadvantages of the candidate concepts are summarized in Figure 4.4-14.

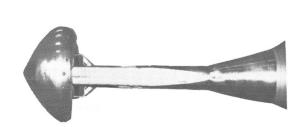
The selection of an auxiliary aerodynamic decelerator requires consideration of a comprehensive set of prerequisites. The selection criteria we used are:

- a. Probability of mission success
- b. System performance
- c. Development risk
- d. Versatility
- e. Cost

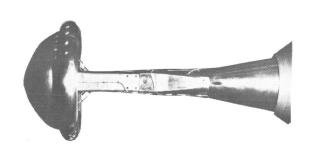
Selection criteria are subdivided into pertinent topics and each candidate evaluated on these bases is shown in Figure 4.4-15.

A major factor in our choice of decelerator is the many years of parachute technology on which one can rely (Reference 4.4-7). The experience gained during the successful history of usage in recovery of spacecraft will promote a less extensive development program, accompanied by less risk and less cost. Although rather severe high frequency oscillations and pulsations have been experienced supersonically and pendular oscillations have characterized descents in low density atmospheres, the recent successes in the PEPP tests are major steps in

ILLUSTRATIONS OF MCDONNELL BALLUTE WIND TUNNEL TEST CONDUCTED AT DOUGLAS TRISONIC 4 FT WIND TUNNEL



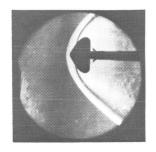
Test S-119 20 July 67 B₂ Unclassified



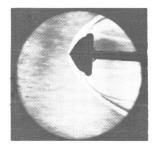
Test S-119 20 July 67 B₂F₂ Unclassified



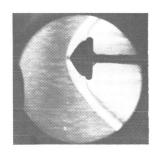
M = 4.0 $\alpha = 0$



M = 2.25 $\alpha = 0$



M = 4.0 $\mathbf{C} = 0$



M = 2.5 $\alpha = 0$

Figure 4.4-13

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TYPE	ADVANTAGES	DISADVANTAGES
Supersonic Ribbon Parachute	 High deployment Mach number, giving more time for separation. 	• Inefficient subsonically
	 Good inflation characteristics at high Mach numbers (up to Mach 3). Ease and compactness in stowing. Good development status. 	• Sensitivity to winds
Supersonic Solid Parachutes	 Feasible at moderate Mach numbers proven in operation. 	 Sensitivity of inflation and sta- bility to forebody configuration, trailing distance, and wake characteristics above Mach 2.
	 Ease and compactness in stowing Lighter than Ballute for same deceleration requirements below Mach 2. 	• Sensitivity to winds
Trailing Ballutes	 Better drag effectiveness than parachutes above Mach 2. 	 Auxiliary inflation probably required (pressure bottles, evaporation, etc.)
	 High Mach number capability, reducing constraints on triggering system. 	
	 Scale model wind tunnel testing easier and more predictable. 	Heavier than attached Ballute
	 Operational capability proven at extremely high altitude and Mach number (M = 9.7, h = 227,000 ft). 	
	 Excellent inflation and trailing characteristics demonstrated over high Mach range in both symmetrical and unsymmetrical wakes. 	• Less drag than attached Ballute
Attached Ballutes	 More stable and less susceptible to atmospheric properties, wake flow and winds than trailing decelerators, resulting in better television coverage. 	 Little experience Difficulty in stowing since aft structure and de-orbit motors prohibit continuous covering of aft surface of Aeroshell. Heavier than parachutes for same
Tucked-Back Ballutes	Stowage around Aeroshell periphery	deceleration requirement. • Little experience
	feasible.	• Little experience
	 Lighter than trailing Ballute. 	
Airmat Cone	 Feasibility proven 	Much heavier than parachutes.

Figure 4.4-14

EVALUATION OF AERODYNAMIC DECELERATORS

	ı			
		SUPERSONIC RIBBON PARACHUTES	SUPERSONIC SOLID PARACHUTES	TRAII
PROBABILITY OF MISSION	System Reliability	Successful operation well-proven below Mach 3.	PEPP Test have proven feasibility up to Mach 1.6.	Successful at high alt
SUCCESS		Reefing (if required) reduces reliability.	Reefing (if required) reduces reliability.	(M = 9.7, h
	Vulnerability to Environmental Uncertainty	High vulnerability because of poor inflation stability above Mach = 2.0 — 2.5	Questionable because of operational feasibility above M = 1.6.	Low becau characteris range.
	Simplicity	Good. Compact packaging. Easily deployed.	Good. Compact packaging. Easily deployed.	Good. Com deployed.
SYSTEM PERFORMANCE	Drag	High subsonic, decreases rapidly between Mach 1.0 and 3.0	High subsonic and transonic super- sonic drag unknown above approx. M = 1.6.	Low subso low supers
	Mach Operational Capability	Acceptable up to about M = 3.	Unknown above M = 1.6	Demonstrat
	Stability	Poor inflation stability. Poor pay- load stability under gusty environ- mental conditions.	Poor inflation stability. Poor pay- load stability under gusty environ- mental conditions.	Excellent payload sto vironmenta
	Weight	Low drag to weight ratio	High drag to weight ratio.	Extremely
DEVELOPMENT RISK	Present State of Development	Good	Dependent on PEPP results. Appears feasible.	Excellent.
	Development Requirements	Complex trailing behavior requires custom development.	Further development required beyond PEPP.	Outstandin Iow develo
	Development Difficulty	Complex trailing behavior can cause problems.	Believed to be average based on PEPP.	Little expe
VERSATILITY	Growth Capability	Limited by custom development and Mach operational character-istics.	Unknown. May be good depending on future developments.	Good due t tional capa
	Sensitivity to Configuration Adaptability	Good. Installation flexible.	Good. Installation flexible.	Good. Inst
COST	Flight Tests	Requires full scale tests.	Requires full scale tests.	Fewer full cause wind system sco
	Wind Tunnel Tests	Past experience indicates full scale testing required. Wind tunnel tests of little use.	Estimated similar to ribbon parachutes.	Useful in s duce cost of costs.
	Testing Difficulty	Can become complex on basis of past experience.	Estimated similar to ribbon parachutes.	Moderate t

Figure 4.4-15

ING BALLUTES	ATTACHED BALLUTES	TUCKED-BACK BALLUTES	AIRMAT CONES
operation well-proven	Reliability yet to be proven.	Reliability yet to be proven.	Reliability yet to be proven.
rude and Mach No. = 227,000 ft)	Initial auxiliary pressurization system required to initiate inflation.	Initial auxiliary pressurization system required to initiate inflation.	Pressurization system required to sustain inflation.
e of good operational tics over wide Mach	Unknown, but should be low, simi- lar to training Ballutes.	Unknown, but should be low, similar to trailing Ballutes.	Unknown, but should be low, similar to trailing Ballutes.
act packaging. Easily	Poor. Complex packaging interferes with other subsystems.	Average. Packaging tolerable, but more complicated by peripheral volume requirements.	Average. Packaging tolerable, but more complicated by peripheral volume requirements.
pic, high transonic, onic.	Estimated good supersonic, poor subsonic.	Estimated good supersonic, poor subsonic.	Estimated good supersonic, poor subsonic.
ed operational capa- M = 10.	Estimated to be similar to trailing Ballutes.	Estimated to be similar to trailing ' Ballutes.	Estimated to be similar to trailing Ballutes.
nflation stability. Poor t bility under gusty en- conditions.	Estimated excellent inflation sta- bility and good payload stability under gusty environmental condi- tions.	Estimated excellent inflation sta- bility and good payload stability under gusty environmental condi- tions.	Estimated excellent inflation sta- bility and good payload stability under gusty environmental condi- tions.
ow drag to weight ratio.	Average drag to weight ratio.	Average drag to weight ratio.	Low drag to weight ratio.
	Unproven. Conceptual stage only.	Unproven. Conceptual stage only.	Feasibility proven.
performance indicates oment effort required.	Extensive.	Extensive.	Extensive.
ited.	Unknown.	Unknown.	Unknown.
large Mach opera- pility.	Unknown. May be good because of similarity to trailing Ballutes	Unknown. May be good because of similarity to trailing Ballutes	Unknown. May be good because of similarity to trailing Ballutes.
llation flexible.	Poor. Little lattitude in installation requirements.	Poor. Little lattitude in installa- tion requirements.	Poor. Little lattitude in installa- tion requirements.
scale tests required be- tunnel testing aids ing.	Unknown. Believed similar to trailing Ballutes.	Unknown. Believed similar to trailing Ballutes.	Unknown. Believed similar to trailing Ballutes.
stem scaling to re- f overall development	Estimated similar to trailing Ballutes.	Estimated similar to trailing Ballutes.	Estimated similar to trailing Ballutes.
low.	Unknown because of unproven concept. Could be complex.	Unknown because of unproven concept. Could be complex.	Unknown because of unproven concept. Could be complex

defining solutions to those problems.

Several points in favor of Ballutes are its higher Mach number operational capability, its greater drag effectiveness above Mach 2, and its reduced sensitivity to wind shears and gusts. Ballutes are also more conductive to scale model wind tunnel testing, and, in the case of attached Ballutes, to dynamic gust predictions since it approaches the rigid, one body analysis. Attached Ballutes would be more stable than either of the trailing decelerator concepts due to the flexible connections of the latter resulting in a two body system which exhibits a more complex dynamic gust response. Ballutes, however, are in relatively early stages of development.

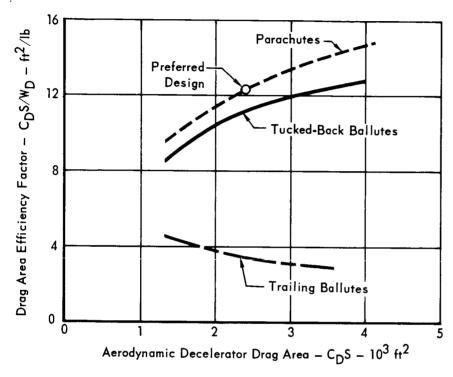
Parachutes are more effective both subsonically and transonically, and result in less weight (on the order of 15-40 lb) than a Ballute for a specific deceleration requirement as shown in Figure 4.4-16. Trailing Ballutes are even heavier than the attached variety for similar deceleration requirements. The large trailing distance associated with parachute deployment is advantageous since wake effects are reduced.

Wake effects are unknown during the separation maneuver with an attached Ballute and may be unfavorable as indicated earlier in the discussion of the separation-reattachment phenomenon. Ballute behavior is also unknown during separation due to its change in shape with the new attachment points. Stowage of the attached Ballute must be accommodated around the periphery of the Aeroshell and would occupy large areas where other hardware attachment points are desired. Large quantities of heat protective coverings would be required with this stowage arrangement, whereas parachutes stow in a compact package and are more easily deployed.

4.4.6 <u>Conclusions and Recommendations</u> — For mission success it is imperative that the aerodynamic decelerator perform its tasks in a reliable and predictable manner. High operational reliability is insured by employing proven concepts in practical systems. Throughout the years of development, parachutes have demonstrated their unquestionable reliability when employed in properly designed systems. Proven principles and conservative engineering practices insure system reliability, as demonstrated by McDonnell's successful history with parachute system applications. This experience aids the decision making processes, and it increases our confidence in the validity of conclusions and decisions concerning system design and operation.

Feasibility is the important factor which must be demonstrated; the NASA PEPP tests have demonstrated parachute feasibility in the operational regime required for

COMPARISON OF DRAG AREA EFFICIENCY FACTOR FOR PARACHUTES AND TUCKED – BACK BALLUTES



W_D = Decelerator System Weight - Ib C_DS = Subsonic Drag Area - ft² Basis: Deployment q = 10-15 psf Opening Loads 15,000-20,000 lb the VOYAGER application. The tucked-back Ballute is the best Ballute-type concept. However, it lacks the important benefit of having proven feasibility. We believe Ballute concepts are in an earlier state of development than parachutes, and additional proof of attached Ballute feasibility is required. This important consideration along with others, leads us to the choice of the supersonic solid parachute as our preferred VOYAGER auxiliary aerodynamic decelerator.

- 4.5 TERMINAL DECELERATOR SELECTION The Aeroshell decelerates the capsule to less than 1000 ft/sec by aerodynamic drag. A combination of an auxiliary aerodynamic decelerator and a terminal propulsive decelerator is selected to augment the deceleration provided by the Aeroshell. The study considered four alternatives: two combinations of aerodynamic decelerators and propulsion subsystems and two alternatives of propulsion only subsystems. The preferred propulsion subsystem was decided in a parallel trade study (See Section 4.3); the preferred aerodynamic decelerator selected is a parachute (See Section 4.4); and the preferred separation technique is differential drag (Section 4.6). This study was limited to the establishment of a configuration of propulsion only or of combined decelerators. selection process includes Ballutes to assure completeness of analysis of combined subsystems. No selection independent from Section 4.4 is inferred. 4.5.1 Summary - The configurations employing a combination of aerodynamic and propulsive decelerators are preferred to all-propulsive configurations. The combination of a 70 ft nominal diameter parachute, and a four-engine, throttleable, bipropellant rocket engine subsystem is the preferred configuration. A 58 ft diameter, tucked-back Ballute could be substituted for the parachute with the same performance at thrust initiation. The least expensive, highest component reliability, but heaviest decelerator investigated is an all-propulsive configuration based on a solid propellant rocket with six monopropellant verniers which fire
- 4.5.2 <u>Functional and Technical Requirements</u> The VOYAGER Capsule Bus is decelerated from entry velocities ranging from 13,000 to 15,000 ft/sec to equilibrium velocities of 265 to 860 ft/sec depending on atmosphere and entry m/C_DA (baseline value = 0.266 slugs/ft²). The requirement for a soft landing, defined as less than 25 ft/sec vertically and 10 ft/sec horizontally, necessitiates use of a terminal propulsion system that can decelerate the Capsule Lander to this condition while controlling its attitude. A terminal propulsion system can be configured to fulfill this requirement and to separate the lander from the Aeroshell. However, the equilibrium velocity specturm is so broad that an auxiliary aerodynamic decelerator can improve performance with high probability of mission success. Therefore, a study was instigated to select a preferred decelerator combination for the 1973 mission which would perform the following functions:

through the Aeroshell.

- a. Decelerate the Capsule Lander from a velocity of over 1000 ft/sec down to less than 25 ft/sec velocity at landing.
- b. Control vehicle attitude throughout the propulsive descent phase.

- c. Separate the Capsule Lander from the Aeroshell with a minimum of recontact possibility and of landing radar interference.
- d. Shut down the propulsion subsystem in a manner that causes no instability at touchdown.
- e. Provide growth capability to 1979 requirements (7,000 lb max Flight Capsule weight).

The Capsule Bus decelerators are required to meet these technical requirements:

- a. Capsule overall weight = 5000 lbs
- b. Capsule entry weight = 3680 1bs
- c. Weight at thrust initiation (Aeroshell and deployable decelerator released) = 2650 lbs
- d. Weight at thrust initiation (propulsion only) = 3680 lbs
- e. Capsule Lander weight at touchdown = 2260 lbs
- f. Ballute deployment, $M \leq 5$
- g. Parachute deployment, M 2
- h. Thrust termination altitude: 10 + 1 ft/sec
- i. Thrust termination velocity: $V_v = 10 \pm 1$ ft/sec, $V_h = 0$ to 1 ft/sec
- j. Verticality: Roll axis ≤ 11° from the vertical, to meet lander subsystem requirements (Section 5.4)
- k. Atmospheres: VM-1 to VM-10
- 1. Entry Conditions: $V_e = 13,000$ to 15,000 ft/sec, $\gamma_e = \text{graze to } -20^\circ$.
- 4.5.3 <u>Design Approaches and Significant Characteristics</u> Several concepts for decelerator configurations have been considered in the trade studies discussed in Sections 4.1 through 4.4 and in 4.6. Of these, certain configurations have appeared to be especially advantageous for the VOYAGER Capsule Bus. Although many were considered for the deceleration task, only these were chosen for final consideration:

	Auxiliary <u>Aerodecelerator</u>	Separation <u>Technique</u>	Propulsive <u>Decelerator</u>
Α.	Supersonic Parachute	Differential Drag	4-engines; canted nozzles
В.	Tucked Back Ballute	Differential Drag	4-engine; canted nozzles
С.		Fire-through-Holes	4 engines; canted nozzles
D.		Fire-in-the-Hole	Single Solid + Six Verniers

In Section 4.3 the 4-engine configuration is selected as the preferred terminal propulsion configuration for the Capsule Lander; in Section 4.4 the supersonically deployed parachute is selected as the preferred deployable aerodynamic decelerator,

and in Section 4.6 the differential drag method of Capsule Lander/Aeroshell separation technique is selected as the preferred confiugration when combined decelerators are employed. With propulsion only, the Capsule was adapted to the propulsion system. These are herein evaluated for significant characteristics, particularly those which may have overriding influence on the Capsule Bus design.

The all-propulsive mode is also evaluated bacause it omits the deployable decelerator (and its deployment mechanisms) and is apparently less complex. The fire-in-the-hole separation technique (D, above) is superior to the fire-through-holes technique (C), but the solid motors high packing density and its ability to decelerate the lander rapidly from supersonic to subsonic speeds was sufficiently attractive to warrant its inclusion in this study.

A preprogrammed type terminal trajectory was stipulated in this study (See Section 4.3). Even the fire-through-holes configuration has a preprogrammed mode switching line and constant velocity descent, to keep the comparison based on similar system dynamics.

4.5.3.1 <u>Significant Characteristics</u> - The significant characteristics or selection factors are deceleration capability, verticality improvement, separation simplicity, wieght optimization, and reliability enhancement. The terminal propulsion system can be sized to handle the entire final deceleration and can also be used for separation. Therefore, the comparison is between improved performance and increased complexity. Improved performance is exemplified by decreased weight of all decelerators and increased verticality (roll axis rotated from -20° to -90° from horizontal). Reliability enhancement is obtained by lowering separation of Aeroshell velocities, lowering landing radar requirements, and shorter burn times. 4.5.3.2 Candidate Concepts - The Capsule enters the Martian atmosphere at hypersonic speeds at a defined altitude of 800,000 feet with negligible deceleration until the Capsule descents to an altitude below 100,000 ft. Deceleration by the Aeroshell alone would result in surface impact velocities from 265 ft/sec to 860 ft/sec depending on the ballistic parameter (m/C_DA) and the atmospheric composition. Trailing decelerators and extended Aeroshells can provide deceleration and stabilization additional to that of the Aeroshell. Impact velocities on the order of 100 to 300 ft/sec can be obtained through the sequential use of Aeroshell and deployable drag devices. However, to achieve the specified 25 ft/sec vertical velocity for soft landing, deployable decelerators with diameters in excess of 200 ft would be required in even the most favorable atmosphere (VM-10). Therefore, aerodynamic decleeration alone will not produce a soft landing. In addition, an

aerodynamic decelerator cannot remove the high ground drift caused by the anticipated winds. In combination with a terminal propulsion system, however, a deployable aerodynamic decelerator will optimize deceleration while increasing the Capsule's susceptibility to wind drift.

The candidate deployable aerodynamic decelerator concepts are:

- o Supersonic solid parachute
- o Tucked-back Ballutes

The candidate terminal propulsion concepts are:

- o Bipropellant-monomethyl hydrozine and nitrogen tetroxide (deceleration plus attitude control)
- Solid propellant ammonium perchlorate (deceleration); monopropellant hydrazine (attitude control).

Characteristics of the four implementations are presented in Figure 4.5-1.
4.5.4 Evaluation - The evaluation proceeded from the establishment of terminal decelerator initial conditions and the operational constraints of the landing radar subsystem to a trade of capabilities. The basic configuration assumptions are given in Figure 4.5-1. Conditions that prevail at the initiation of auxiliary deceleration are tabulated along with engine configurations, relationships and sequence of events. Comments relative to the guidance and control aspects are included to characterize each configuration.

4.5.4.1 <u>Probability of Mission Success</u> - The probability of the Aeroshell and the auxiliary decelerators successfully slowing the Capsule Lander to required touchdown velocities is a function of component reliability, operation complexity, and extremes of environment. Component and subsystem reliability estimates were based on the nature of the hardware and on duration of operation. Engine operation duration is decreased when propulsive deceleration is used in conjunction with a parachute or Ballute. (Section A2.3.7). However, the times are not dramatically different, because the slower average velocities after the aerodynamic deceleration increments have occurred, result in roughly equivalent operation times. Operation is illustrated in Figures 4.5-2 and 4.5-3.

Assumed Terminal Propulsion Subsystem Burn Times

0	Parachute with	four	bipropellant e	engines:	50 sec	(A)

o Ballute with four bipropellant engines: 50 sec (B)

- o All propulsive four bipropellant engines: 70 sec (C)
- o All propulsive solid motor plus six monopropellant (D)

verniers: 60 sec

SUMMARY OF DECELERATOR CONFIGURATIONS

	CONFIGURATION A DIFFERENTIAL DRAG, PARACHUTE, FOUR BIPROPELLANT ENGINES	CONFIGURATION B DIFFERENTIAL DRAG, BALLUTE, FOUR BIPROPELLANT ENGINES	CONFIGURATION C FIRE THROUGH HOLES 4 ENGINES	CONFIGURATION D FIRE-IN-THE HOLE ONE SOLID& SIX MONOPROPELLANT
Basic Assumptions	Aeroshell-Lander separation accomplished by large differential drag provided by a 70 ft parachute. Lander Radar has no capability to see through Aeroshell. Landing Radar angular limitations similar to existing LM System.	 Aeroshell-Lander separation accomplished by large differential drag provided by a 70 ft parachute. Lander Radar has no capability to see through Aeroshell. Landing Radar angular limitations similar to existing LM System 	 Terminal engines fire through Aeroshell Landing radar has capability to achieve acquisition through Aeroshell. Landing radar angular limitations similar to existing LM system. 	 Lander separation accomplished by firing terminal engines and backing away from Aeroshell. Landing radar has no capability to see through Aeroshell. Landing radar angular limitations similar to existing LM system.
Range of Initial Condi- tions Includ- ing Wind Effects)	• 5,000 ft altitude • 100- 405 ft sec vertical vel • · 30 deg sec pitch or yaw rates • Unknown roll rate (10 deg sec assumed) • 0 - · 50 deg altitude from relative wind. 75-90 deg flight path angle	• 5000 ft altitude • 100×405 ft sec vertical vel • ±30 deg, sec pitch or yaw • rates Unknown roll rate (5 deg sec assumed) • 0 ± 50 deg altitude from relative wind 75–90 deg flight path angle	• 15,000 ft altitude • 375 · 1320 ft/sec vertical vel • -23 · -90 deg flight path • : 35 deg sec pitch or yaw rates • : 2 deg sec roll rate • 0 · : 15 deg attitude from relative wind	• 15,000 ft altitude • 375 · 1320 ft/sec vertical vel. • -23 · -90 deg flight path • : 35 deg sec pitch or yaw rates • : 2 deg sec roll rate • 0 : 15 deg attitude from relative wind
Engine Con- figuration and Maximum Impulse Requirements	• 4 Bipropellant engines • (throttlable) 70,000 lb-sec impulse	 4 Bipropellant Engines (†hrottlable) 70,000 lb-sec Impulse 	 4 Bipropellant Engines (throttlable) 170,000 lb-sec Impulse 	 6 Monopropellant Engines (throttlable) + solid rocket 140,000 lb-sec Impulse
Aeroshell. Lander Relationship Following Separation	Based on 70 ft parachute deployed at 23K ft altitude and Aeroshell separation 12 seconds later, the shell will impact with surface prior to radar search.	 Based on 58 ft diameter (projected) ballute deployed at 20K ft altitude and Aero- shell separation 8 seconds later, the shell to impact prior to radar search. 	• Aeroshell retained until final constant velocity descent phase. Aeroshell released and Lander performs lateral maneuver Radar not utilized during this final phase.	• Lander passes Aeroshell 2 to 15 sec following ignition assuming a 200 ft sec ² acceleration capability and a 1 sec burn.
Basic Sequence	• Terminal Engines ignited at 5,000 ft altitude. After large	 Terminal Engines ignited at 10,000 ft altitude. After large 	• 2 g Thrust ignition • Damp initial rates	 Operational sequence does not appear feasible because

Figure 4.5-1

4-124 -/

	To miss Aeroshell.		
	quired during final descent		
	Additional maneuvers re-		,
	on radar acquisition.	establishing ignition mark.	establishing ignition mark.
	Critical sequence dependent	ingte godinst Aeroshell in	minate against Aeroshell in
	fransmit and receive through	• artitude oscillations.	Altimeter required to discri-
	• Requires landing radar to	that can tilter possible large	That can filter possible large
	acquisition.	attitude reference at ignition	attitude reference at ignition
	8) prevent poor velocity	Requires means to establish	Requires means to establish
יים מיים מיים מיים מיים מיים מיים מיים	shallow flight paths (VM-7,	thruster size increase.	thruster size increase
Would require non-standard	Non-standard sequence of scent required because	Least impulse required.	• Growth potential without
		thrusting.	thrusting.
quired to separate from	sequence highly dependent	not dependent on engine	not dependent on engine
 Extra propulsive kick re- 	ficient. Feasibility of above	 Aeroshell-Lander separation II 	 Aeroshell-Lander separation
able influence on response.	applicable to moment coef-	atmosphere.	atmosphere.
	ficient by JPL and assumed	which are standard for any	which are standard for any
Approduction of paracteristics.	determined for drag coef-	 Simple sequence of events 	• Simple sequence of events
dynamics dependent on igni-	tactor of 4 due to thruster	 ot low dynamic pressure 	of low dynamic pressure
• Aeroshell-Lander separation	coefficients decreased by	to cause problems because	ly to cause problems because
dynamics occurs at supersonic	cluded in analysis, however,	phase neglected. Least likely	phase neglected. Least like-
Aeroshell-Lander separation	of Aeroshell-Lander in-	of Lander during powered	of Lander during powered
		-	
	to obtain lateral separation Engine cutoff initiated by feel		
	then to -30 deg and back to		
	cent phase.		
	to the constant velocity des-	altitude.	altitude.
	and subsequent deceleration	and engine cutoff at a set	and engine cutoff at a set
	 Intersection of the preprog- rammed deceleration profile 	descent phase. Constant velocity descent	• Constant velocity descent
	velocity control initiated.	to the constant velocity	to the constant velocity des-
	• At radar acquisition, lateral	 and subsequent deceleration 	and subsequent deceleration
	vertical initiated.	grammed descent profile	rammed descent profile
	down maneuver towards	Intersection of the prepro-	 Intersection of the preprog-
hence areater throttle ratio	1.5 g's and 10 deg sec pitch	• velocity vector.	trolled to velocity vector.
donination greater engine	Durn for approximately 5 sec	and thrust axis controlled to	tion and thrust axis con-
delay would create a re-	After fixed time, max. thrust	tiated K seconds from ignition	tigted 6 seconds from ignition
fire through hole, however,	s initiated		landing radar acquires lock.
events would be similar to	euver towards horizontal is	Initial rates damped and	 Initial rates damped and
shell to clear radar beams.	does not occur affer 3 sec 10 dea sec pitch up man-		less than 1 qM
ot required delay for Aero-		_•	ignition, engines throttled to
المراجات المراجات المراجات المراجات	Badar sparch if acquisition	thrust command to assure	thrust command to assure

Figure 4.5-3 TERMINAL DESCENT TRAJECTORY IN VM-10 ATMOSPHERE 1600 Lateral Control Started Switching Line 1200 Parachute Release -High Thrust Separation & Thrust Initiation Lander & Parachute =Parachute Deceleration Total Velocity - ft/sec Aeroshell Release VM-10 Atmosphere Parachute Deployed Aeroshell Release 8gM Descent 8 Altitude – 10³ ft. 5 20 24 Figure 4.5-2 TERMINAL DESCENT TRAJECTORY IN VM-7 ATMOSPHERE 1600 Aeroshell Only Aeroshell Release Switching Line -.8gM Descent High Thrust Separation -Aeroshell Only 1200 Total Velocity - ft/sec Parachute Deployed-VM-7 Atmosphere Parachute Deceleration— Release 🕂 Initiation Aerosheil Release Parachute & Thrust Lander & Parachute-Lateral 🞝 Control Started 9 ಣ 8 œ 24 Altitude – 10^3 ft

Failure rate for monopropellant throttling is one-half that of bipropellant throttling when control circuitry is excluded. For mission success, all of the monopropellant vernier engines are required to operate in the solid/monopropellant configuration (D).

Reliability Estimates - The estimated reliability of the solid/monopropellant configuration (D) is slightly superior to the other configurations as seen in Figure 4.5-4. The differences are modest, so in view of the recommendation of the trade studies discussed in Section 4.3 and 4.6, the configuration is not recommended for VOYAGER.

The difference in reliability between the configurations having auxiliary aerodynamic decelerators and the remaining all-propulsive configuration is also not very large, but, significantly, the combined deceleration configurations are estimated to be more reliable. The value assumed for parachutes and Ballutes is as yet unsubstantiated. Considerable experience with subsonic parachutes and even some experience with supersonic parachutes does not permit rigorous estimation of this subsystem's value. Similarly, the various high Mach number tests of Ballutes – usually trailing – also are insufficient in number of pertinence to warrant much confidence in any reliability value. Thus, the estimate is largely for the deployment mechanisms, for the inflation techniques, and for the release mechanisms for parachutes. Although these differ in details, they are equivalent in complexity for both parachutes and Ballutes.

Operational Complexity – Operations are complex because of the number of steps imposed and because of contingencies encountered or nonstandard maneuvers required to tolerate a wide band of conditions. For example, an entry at 13,000 ft/sec and at $\gamma_E = -20^{\circ}$ into a VM-7 atmosphere produces high velocities and low flight path angles down to the surface, assuming no auxiliary deceleration. In a VM-10 atmosphere the trajectory is almost vertical and the speeds are slow. The problem in the former case is separation, deceleration, and verticalization. In the latter case, the problem is separation, avoidance of radar lock-on to the Aeroshell, and wind drift (on parachutes). To tolerate extremes of this sort requires, in the case of the all-propulsive lander, torquing maneuvers in the pitch plane that provide extra reduction of the lateral velocity vector to lessen radar acquisition difficulty.

Such nonstandard steps are not required if an auxiliary aerodynamic decelerator is used to funnel the trajectories down to an easily planned for set of initial terminal propulsion conditions. Though not quantified, the selection of the

WEIGHT COMPARISON CHART

CONFIGURATION*	Α	В	U	D
Áerodynamic Decelerator	190	210	-	1
Terminal Propulsion	570	570	990	890
Total	760	780	990	890

RELIABILITY COMPARISON

SUBSYSTEM	
a) Bipropellant Feedb) Bipropellant Engines (4 for 50 sec)c) Bipropellant Engines (4 for 70 sec)	.99647 .97725 .96869
Engine subsystem for combined (a & b) Engine subsystem for propulsion only (a & c)	.97380 .96527
d) Monopropellant Feed e) Monopropellant Engines (6 for 60 sec) f) Solid Rocket Motor	.99723 .97959 .995
Engine subsystem for propulsion only (d,e,f)	.97199
g) Parachute or Ballute	.996
CONFIGURATION	
All Propulsive — Solid/Monopropellant (D) Aerodynamic + Propulsive Decelerators (A, B) All Propulsive — Bipropellant (C)	.97 199 .96990 .96527

^{*}A Parachute + 4 Bipropellant Engines

B Ballute + 4 Bipropellant Engines

C 4 Bipropellant Engines

D 1 Solid + 6 Monopropellant Verniers

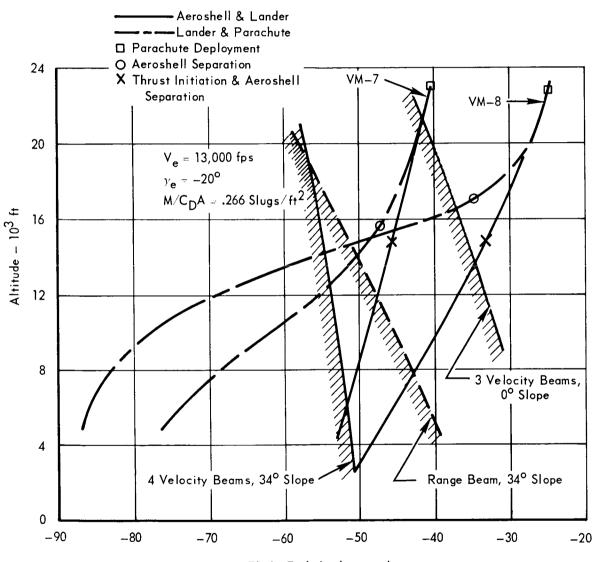
deceleration system and its sizing must consider the effect of a standardized operation sequence on probability of mission success. Those configurations that employ an auxiliary aerodynamic decelerator do enhance probability of success by standardizing sequences.

Similarly, the funneling of parameters down to a small range of variables narrows environmental extremes. Two of the undesirable environments were supersonic separation of Aeroshell and Capsule Lander and thrusting forward into a supersonic flow field. Both are avoided by the use of an aerodynamic decelerator. The development test program accompanying a concept employing fire-through-holes would be extensive. Again, the problem is not so much a question of infeasibility of the all-propulsive concepts as the fact that approaches were available which had no comparable combinations of difficult simulations. The current Planetary Entry Parachute Program is performing tests in which sequencing comparable in complexity to that required in simulation tests of the four configurations; however, the added environments of Saturn V launch, extended cruise, and orbit insertion environments are absent. The addition of supersonic Aeroshell/Lander separation to PEPP would stretch it out. Any reduction in severity of environments equates to higher probability of success.

Canopies (parachutes) and attached or tucked-back ballutes produce different environments due to their different physical response to gusts, shears, and opening loads. The differences are not conclusive at this point. See Sections 4.3 and 5.10 for further discussion of operation capabilities of the aerodynamic decelerators.

Landing Radar Limits - The feasibility of operating the landing radar is a function of inherent design, altitude, and flight path angle. The design was assumed to be fixed for this study. The radar has four velocity beams having a squint angle of 20°; only three signals are required for the landing radar to control system operation. A complete discussion of the radar is presented in Section 5.9 but its limits do influence capability needed in the decelerators. The primary limit of interest is that which applies to three (of four) velocity beams and a 0° ground slope (See Figure 4.5-5). The beams are considered to be in the worst roll condition (two beams below, two above the yaw plane). As seen, the separation of the all-propulsive configuration is defined as above this limit. During separation, and for up to 15 seconds after Aeroshell separation, the Capsule Lander is under attitude hold control conditions so that the definition is appropriate. However, the altitude of 15,000 ft was selected because of the time required in all

LANDING RADAR LIMIT CURVES



Flight Path Angle, γ – deg

Figure 4.5-5

4-129

atmospheres to perform the functions of separation, of delay until Aeroshell impact, and of minimization of lateral velocity which dictate a high altitude separation. Thus, the definition has little or no margin for extreme environment operation. This is a situation where the threshold of infeasibility is close, thereby making these systems low value candidates.

Thus, there is a numerical difference in estimated reliability favoring an all-propulsive configuration based on a solid rocket motor and six monopropellant vernier engines. Both combination configurations are next in reliability. All-propulsive configurations are last, since they operate at or near landing radar limits. Thus, the combined aerodynamic and propulsive decelerator configuration is preferred because of the standard sequencing can be used and better control of operational environment at the initiation of terminal propulsion can be achieved. 4.5.4.2 Performance - System performance of decelerators is demonstrated by the amount of weight that must be allocated to perform the required function of slowing down the lander. The weights of the two types of decelerators are 190 lb and 210 lb as given in Figure 4.5-4. The value quoted for a Ballute is conservative. Design conditions can be set which could reduce this to a value equal to that of a parachute. The conservative estimate is included because there is to date no practical development demonstration of inflatability of an attached Ballute under simulated Martian conditions.

The weight of the two combined systems is low compared to the all-propulsive systems. The 130 lb increase is sufficient to reduce the Capsule Bus weight contingency and could be considered as an overriding influence. In the interest of completeness, however, the systems were evaluated on other bases as well. Figure 4.5-6 depicts some of the characteristics of the rocket engines considered in this analysis. These differ in some respects from the values in Section 4.3 and are more representative of the preferred Capsule Bus design.

Further weight optimization is possible in the combined systems. The design value for terminal propulsion total impulse is 70,000 lbs-sec which is virtually at the minimum required value relative to initiation velocity at 5000 ft. The parachute, as discussed in Section 4.4, is sized to a compromise of separation differential drag (minimum size), rotation of the combination in high horizontal velocity winds (maximum size) separation loads, delay to permit Aeroshell impact, and packaging volume. System performance can be improved by reducing parachute diameter - thereby permitting more rapid descent velocities - as indicated by overall weight of the Capsule Lander. The gain, however, is small because only

CHARACTERISTICS OF CANDIDATE TERMINAL DECELERATION CONCEPTS

THRUST IMPULSE WEIGHT VOLUME EXHAUST INTERACTIONS (LB) (FT ³)	Base Recirculation Effects -	1650/ 70,000 548 7 85 Base total pressure: 0.0032 psia			1650/ 110,000 752 11.46 thermal coatings.	Engine Base Recirculation Effects same as first)	Possible contamination of optical transparencies or thermal coatings	2	Solid: Solid: Engine exhaust jets firing counter to supersonic free stream could create tipping moments as a result	990 19.34 of uneven pressure distributions on the Aeroshell. Moveable nozzle	Verniers: Covers are required to minimize 450/ 27,900 blockage from Aeroshell products	Although data is not available, re-	be more severe than for the bipro-
						9 5 5 5 1			 		::		
SEPARATION MODE	Differential Drag:		Supersonic	5	Tucked Back	, a	Fire through the	<u>•</u>	Aeroshell retained		Fire in the hole		
MECHANIZATION	4 Bipropellant Engines; Canted	Nozzles	Auxiliary Aerodynamic	Decelerator	4 Bipropellant Engines; Canted Nozzles	Accoloration Decelerator	4 Bipropellant Engines; Canted Nozzles	All Propulsive Deceleration	Solid Motor; 6 Monopropel-	All Propulsive	Decelerator		

Figure 4.5-6

parachute subsystem weight is reduced. Hence, the preferred configuration is near the optimum size insofar as performance is concerned.

4.5.4.3 <u>Development Risk</u> - Development risk is defined as high if a particular development program is estimated to equal or to exceed the available time. Risk is also high if two programs of two items must be done in series and both items must be successfully developed. Similarly, if two items such as decelerators are to be developed concurrently and both must work in series, there is an element of high risk unless one or the other or both can, within the time span allotted, be resized to do the whole job. In this case the items acquire an aura of functional redundancy, especially their development programs.

Because parachutes are currently being proven feasible, the development risk of parachute subsystems is judged to be low. Similarly, since some wind tunnel air testing has been done and some high altitude simulation testing of Ballutes may soon be instituted, the risk in a Ballute development is judged to be higher than that for parachutes but not significantly so.

The thrust of the propulsion subsystem for touchdown control, subsequent to descent on the auxiliary aerodynamic decelerator, must be sized on the same basis as that for an all-propulsive decelerator except for tankage sizes. The minimum thrust requirement is established by the minimum weight and the minimum preprogrammed acceleration. Thus, the least thrust is required in the 1973 mission when the Capsule Lander is least in weight. The maximum thrust condition is a function of either the maximum practical throttling ratio, the maximum thrust condition in the least favorable atmosphere currently defined (1973), or the maximum thrust condition required in the later, heavier missions. Atmospherics, as such, do not directly affect the maximum value; their effect is indirect. The low density atmospheres yield high velocities which inhibit the time available to perform all functions. Therefore, since duration is not appreciably different, it is possible that if the development of auxiliary decelerators should prove exceedingly difficult, redirection to an all-propulsive configuration would occur.

Thus, the development of both aerodynamic and propulsive decelerators is a risk because each is estimated to take a major portion of the time available for the 1973 opportunity. The developments are not end-to-end, but, since the aerodynamic decelerator end conditions establish the beginning conditions of the propulsive one, the risk involved in a combined system is higher than for either alone.

4.5.4.4 Versatility - The ability to change plans for entry and deceleration mode between the two Capsule Busses, if desired, is facilitated by the combination system.

An all-propulsive configuration will have some limited versatility because the design is sized to work over a large entry window and for a wide spectrum of atmospheres. However, the capability of using different regimes of entry velocities and altitudes is more available for a combined system than an all propulsive one. For example, the limited excess deceleration capability available from the parachute (see 4.5.4.2) provides some growth to later, heavier missions. 4.5.4.5 Costs - The costs of the combination systems are estimated to be essentially the same: \$97.6 million for parachute and rockets, and \$97.7 million for Ballutes and rockets. The all-propulisve systems are both less: \$81.5 and \$52.9 million for the four engine and the solid/monopropellant configurations, respectively. The latter assumes the suitability of an existing solid rocket motor. fore, the all-propulsive systems are less costly than combined systems. 4.5.5 Recommendations - The combined decelerators are superior to the all-propulsive configurations because they have a higher probability of mission success, performance, and versatility. They pose a higher development risk and they cost more, however. On the basis of this, a combination of an auxiliary aerodynamic decelerator and a propulsive terminal deceleration subsystem is recommended - in particular, one having a parachute as the deployable decelerator. (See Figure 4.5-7). tucked-back Ballute is a high value alternate deserving continued consideration.

COMPARISON OF DECELERATOR CONFIGURATIONS

Figure 4.5-7

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- 4.6 AEROSHELL/LANDER SEPARATION This trade study was performed to determine the optimum means of separating the Aeroshell from the lander and to provide needed information for selection of the best separation altitude. The resulting decision has an important interrelated effect on the lander design concept, Aeroshell design, propulsion subsystem, guidance and control subsystem, auxiliary aerodynamic decelerator, landing radar design, descent science, telecommunications subsystem and system reliability. This study was made concurrently with those of the terminal deceleration method (See Section 4.5) and choice of aerodynamic decelerator (See Section 4.4).
- 4.6.1 <u>Summary</u> The study was conducted in two phases. Thirteen candidate separation concepts were evaluated in the preliminary phase. Four candidate concepts were evaluated in more depth during the final phase of the study. The differential drag concept was then selected as the preferred method of separation. A parachute is used as the auxiliary aerodynamic decelerator with deployment at 23,000 feet and Aeroshell separation at a fixed time delay of 12 seconds thereafter.
- 4.6.2 <u>Functional and Technical Requirements</u> Selection of a preferred separation technique was complicated by the large number of possible candidate approaches to be considered and the varied requirements imposed. The necessity for functioning without interfering with other subsystems was primary among the latter.
- 4.6.2.1 <u>Safe and Successful Separation</u> Obviously, the Aeroshell and lander must physically separate in a positive and safe manner with a high degree of reliability. The possibility of any physical recontact between the lander and the Aeroshell (either whole or with any part) must be minimized.
- 4.6.2.2 <u>Radar Interference</u> A primary concern in many of the candidate concepts was the possibility that the landing radar would "lock-on" to the separated Aeroshell, thus interfering with the normal terminal descent and possibly creating a disastrous malfunction. Another radar consideration, in the event of delayed separation,

was the requirement that the radar operate through the Aeroshell, creating the need for a special radome and complicating the entire landing radar subsystem design. 4.6.2.3 Atmospheric Uncertainty - The separation method must be capable of operating throughout the entire range of possible atmospheres. Figure 4.6-1 shows the range of pre-separation trajectories for the Capsule with and without an auxiliary aerodynamic decelerator (in this case, a 30 foot Ballute). This figure illustrates the wide range of separation altitudes and velocities that must be considered in the selection. While excessive velocity at separation is not desirable, neither is too low a velocity, since this, combined with a maximum wind, would create a flight path angle too close to the horizontal.

- 4.6.2.4 <u>Time to Terminal Descent</u> The selected approach must allow sufficient time for all the functions of terminal descent to take place including radar lockon, terminal propulsion ignition, and all events of the programmed deceleration sequence.
- 4.6.2.5 <u>Interference with Subsystem Operation</u> The preferred separation technique should insure minimum interference with the operation of other subsystems both during and after the separation event. This includes Guidance and Control, Telecommunications and Entry Science.
- 4.6.2.6 <u>Minimum Weight</u> A major consideration in the selection is the potential weight penalty imposed by the candidates. In each case, the total terminal propulsion impulse required has a distinct effect on the weight required.
- 4.6.3 <u>Design Approaches and Significant Characteristics</u> At the outset of this study thirteen candidate concepts were considered for evaluation. Some of these candidates were variations of others but the differences were considered significant enough to warrant separate evaluation. To deal with the large number of alternatives and the many considerations involved, the study was divided into two phases. The preliminary phase had as its objective the reduction of the number of candidates to a workable few and resulted in the elimination of all but four concepts. These four were then analyzed in some depth during the final phase of the study, with emphasis placed on those areas which had been defined as problem areas during the preliminary phase. The original thirteen candidates are described below. The four concepts which were retained through the final phase are asterisked. Figure 4.6-2 shows the original thirteen candidates and some of the major factors which affected their rejection or retention.
- *4.6.3.1 <u>Fire-in-the-Hole (Multiple Rockets)</u> The terminal propulsion subsystem, consisting of three or more throttleable rocket engines is used to decelerate the

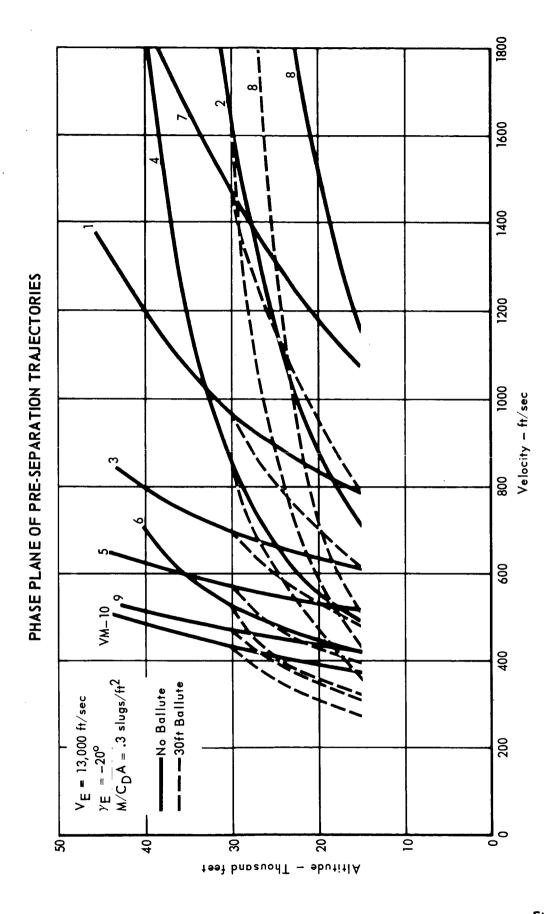


Figure 4.6-1 4-137

COMPARISON OF CONCEPTS - AEROSHELL SEPARATION - PRELIMINARY PHASE

					
	(4.7.4.1) FIRE-IN-HOLE, MULTIPLE ROCKETS	(4 .7.4.2) FIRE-THRU-HOLE, DELAYED SEPARATION	(4.7.4.3) LANDER THRU NOSE, CAP RETAINED	(4.7.4.4) LANDER THRU NOSE, CAP FRACTURED	(4.7.4.5) TRACTOR ROC EXIT AFT
Probability of Successful Separation	Some uncertainty in aeroshell dynamics. Simple and reliable.	Rocket plumes will cause changes in aeroshell drag and stability.	Use of drag device to remove aeroshell is attractive. Nose section severance complex.	Use of drag device to remove aeroshell is attractive. Nose section fracturing complex.	Separation dynamicomplicated.
Unlikelihood of Physical or Radar Recontact	Radar lock-on appears avoidable. Physical recontact appears avoidable.	No radar lock-on problem. Requires programmed lateral movement to avoid physical recontact.	No radar lock-on problem.	No radar lock-on problem. Pieces of cap may strike Lander	Radar lock-on appears avoidab
Absence of Developmental Problems	Some wind tunnel testing necessary.	Extensive wind tun- nel testing probably necessary for jet effects. Radome and radar development probably required.	Wind tunnel testing necessary. Pyrotechnics needed to function after exposure to entry temperatures of 500° to 800°F.	Wind tunnel testing necessary. Pyrotechnics needed to function after exposure to entry temperature of 500° to 800°F.	Wind tunnel test for aerodynamics during separatio very complicated
Lack of Interference with Other Subsystems	Requires no unusual sensors with stringent dynamic range requirements.	Delayed separation enhances entry science accumula- tion. No unusual sensors required.	Separation shock quite severe.	Separation shock quite severe.	
Absence of Thermal or Contamination Problems	Experiment optical contamination due to plume recirculation. Severe heating from rocket exhaust gas.	Some rocket exhaust heating during separation.	Few thermal problems. No contamination.	Few thermal prob- lems. No contami- nation.	Experiment optic contamination du to plume recircu Heating from roc exhaust.
Simplicity of Design	Requires thermal shield to protect Lander.	Porting for rockets complicates design.	Pyrotechnics com- plicate design. Method sensitive to Lander geometry.	Pyrotechnics com- plicate design. Method sensitive to lander geometry.	Tractor rocket packaging comp Requires some heat shielding.
Weight Economy	$\Delta WT = 50^{+50}_{-50}$	$\Delta WT = 50 + 50 - 50$	$\Delta WT = 225 + 75 - 125$	$\Delta WT = 200 + 100 - 100$	ΔWT = 190 +1

Figure 4.6-2

4-138-1

			l .	
ıl ; ıtion. et	Worst case contamination and heating problems for experiment integration. Severe heating from rocket exhaust gas.	Few thermal problems. No contamination.	Lower altitude deployment — less severe heating. No thermal problems.	Lower altitude deployment — le severe heating. thermal problem
ex.	Requires thernal shield to protect lander, otherwise simple.	Requires rotation device. Requires computer and memory for attitude control.	Requires no rockets. Simple.	Requires no roc Simple.
0 0	$\Delta WT = 50 + 50$ -50	$\Delta WT = 150 + 40 - 10$	$\Delta WT = 0 + 130$ -100	Δ WT = 400 $\stackrel{+}{-}$
			1 126	_

(4.7.4.6)

FIRE-IN-HOLE,

(1) LARGE

ROCKET

Some uncertainty

dynamics. Simple

appears avoidable.

Some wind tunnel

Single rocket

interferes with

field of view.

radar location and

testing necessary.

in aeroshell

and reliable.

Radar lock-on

ET,

ics

١g

(4.7.4.7)

RELEASE

AEROSHELL

Rotation requires

ballute or other

dynamic force.

No radar lock-on

Complex wind

tunnel testing.

Radar and camera

recovering stable

attitude. Loss of

during separation

negates all science

experiments during rotation and restart.

communication

time lost while

problem.

ROTATE CAPSULE,

t - less ating. No oblems.

(4.7.4.9)

TRAILING

BALLUTE

OR PARACHUTE

Simple and therefore

Radar lock-on may be

avoidable. Physical

Wind runnel testing

complex. Full scale

testing easier than

Same as tucked-back

dynamic environment

but more severe

re-contact highly

reliable.

unlikely.

most.

(4.7.4.8)

TUCKED-BACK

BALLUTE

Simple and therefore

Radar lock-on may

Physical re-contact

be avoidable.

highly unlikely.

45 ft. ballute -

requires complex

Full scale tests

easiest of all.

separation.

wind tunnel testing.

Requires large gyro

angle memory during

reliable.

(4.

LAN

AER

RE'

Same jet

testing (

Radome

develop

Access

difficult

sensors

Restrict

for imag

required

moveme

Landing

with abl

Large the

atter lar

Porting

complic

 ΔWT

no rockets.

+420

-330

4-138-2

7.4.10) D WITH OSHELL AINED	(4.7.4.11) SPLIT AEROSHELL, REMOVE WITH BALLUTE	(4.7.4.12) SPLIT AEROSHELL, PIECES FLY OFF LOOSE	(4.7.4.13) LANDER THRU NOSE, LEGS STORED FORWARD
	Difficult to predict aerodynamic behavior of flapping pieces. Segmentation is somewhat complex.	Large uncertainty in loose pieces of aero- shell. Segmentation is somewhat complex.	Somewhat complicated design, therefore less reliable.
	No radar lock-on problem.	No radar lock-on problem. Pieces of aeroshell may strike Lander.	No radar lock-on problem.
effects as 3.2.2. and radar ment.	Extensive full-scale testing probably required. Pyrotechnics needed to function after exposure to entry temperature of 500° to 800°F.	Extensive full-scale testing probably required. Pyrotechnics needed to function after exposure to entry temperature of 500° to 800°F.	Wind tunnel testing necessary. Pyrotechnics needed to function after exposure to entry temperature of 500° to 800°F.
to soil surface. No unusual required. s field of viewing. Control for wind at on surface.	Separation shock quite severe.	Separation shock quite severe.	Separation shock quite severe,
area strewn tion''crumbs.'' rmal problem ting.	Few thermal problems. No contamination.	Few thermal problems. No contamination.	Few thermal prob- lems. No contami- nation.
for rockets ates design.	Pyrotechnics complicate design.	Pyrotechnics complicate design.	Somewhat complex design for leg clear-ance, Method sensitive to Lander geometry.
240 + 50 -50	$\Delta WT = 225 + 75 - 125$	$\Delta WT = 225 + 75 - 125$	Δ WT = 225 +75 -125

lander by firing at the back side of the Aeroshell with a maximum thrust burst, thus separating the Aeroshell and the lander.

- *4.6.3.2 <u>Fire-Through-Hole, Delayed Separation</u> The terminal propulsion system fires through ports in the Aeroshell. The Aeroshell is retained till a very low altitude (100 feet or less) is reached. This concept and that described in Section 4.6.3.10 are the only ones which have no auziliary aerodynamic decelerator.
- 4.6.3.3 <u>Lander-Through-Nose</u>, <u>Cap Retained</u> The lander drops through a hole in the nose of the Aeroshell, created by pyrotechnic segmentation. The nose cap is carried with the lander and separated prior to landing.
- 4.6.3.4 <u>Lander-Through-Nose</u>, <u>Cap Fractured</u> This concept is a variation of 4.6.3.3. The lander drops through a hole in the nose of the Aeroshell which is created by pyrotechnically fracturing the cap in several (three or four) sections. The sections fall freely away.
- 4.6.3.5 <u>Tractor Motor, Exit Aft</u> Separation is accomplished by a tractor-type rocket that pulls the lander aft. The tractor rocket is erected on some form of tower by mechanical means, possibly using its own traction.
- 4.6.3.6 <u>Fire-in-the-Hole</u>, <u>Single Large Rocket</u> This is a variation of 4.6.3.1 using a single large retrorocket. Exit of the lander is again out the back.
- 4.6.3.7 <u>Rotate Capsule, Release Aeroshell</u> The entire capsule is rotated about the pitch or yaw axis so as to place the Aeroshell above the lander and the Aeroshell is then released. The lander then falls away. A small parachute can be used to rotate the capsule.
- *4.6.3.8 <u>Differential Drag (Tucked-Back Ballute)</u> A tucked-back Ballute is used for transonic stability and deceleration and for subsonic separation. The Ballute, being attached only to the lander, provides a substantial difference in drag area, thus accomplishing separation. (This candidate was studied as typical of those using differential drag. The aerodynamic decelerator ultimately selected is to be used as the drag device.)
- 4.6.3.9 <u>Differential Drag (Trailing Ballute or Parachute)</u> A trailing Ballute or parachute is used for subsonic separation. This is a variation of 4.6.3.8. (This candidate, using the parachute, was ultimately selected.)
- 4.6.3.10 <u>Aeroshell Retained</u> Landing is accomplished with the Aeroshell retained *4.6.3.11 <u>Split Aeroshell</u>, <u>Remove with Differential Drag</u> The entire Aeroshell is pyrotechnically or mechanically split into a number of segments (three or more) at separation. The segments are attached to the differential drag device and when the lander is released it falls and the Aeroshell segments are carried away above

the lander.

- 4.6.3.12 <u>Split Aeroshell</u>, <u>Segments Loose</u> The entire Aeroshell is pyrotechnically or mechanically split into a number of segments which are allowed to fly off freely.
- 4.6.3.13 Lander Through Nose, Legs Stored Forward of Prime Structure (Note: At the time this study took place our baseline lander concept contained legs.) This separation concept is a variation of 4.6.3.3 and 4.6.3.4. The lander drops through the nose but the legs (which are large and difficult to clear) clear the Aeroshell by being stored forward of the basic Aeroshell structure. The legs are protected only by ablative covers while stored.
- 4.6.4 <u>Description of Analyses and Studies</u> Two purposes were paramount in this study; first, to determine feasibility, and second, to develop basic information needed for design iteration.
- 4.6.4.1 <u>Trajectory Studies</u> From the outset of this trade study, the possibility of radar lock-on to the separated Aeroshell or of physical recontact with the separated Aeroshell was among the most serious potential problems. This was particularly true for the Fire-in-the-Hole and the Differential Drag (tucked-back Ballute) concepts and was equally applicable to the use of a parachute for separation. It was most severe for the high density atmospheres where the flight path is essentially vertical at separation. Following a straight separation out the rear, the Aeroshell would be directly in the path of the landing radar range beam and depending on the separation velocity, the two bodies might even collide. In order to investigate the severity of this hazard, a digital computer simulation was written. The results are pertinent in comparing the Fire-in-the-Hole concept with the others. The simulation was based upon the following constraints and assumptions:
 - o Motion was restricted to rotation and translations in the entry phase.
 - o The Aeroshell was considered to be a point mass; i.e., drag effects only.
 - o The aerodynamic forces and moments on the lander were considered negligible.
 - o A stylized control system which performed perfect deceleration throttling with no system lags was assumed. A first order attitude control was assumed which incorporated attitude rate limiting.
 - o The flight path of the Aeroshell was not perturbed during the separation impulse.
 - o The Aeroshell was assumed to be aligned with its flight path.

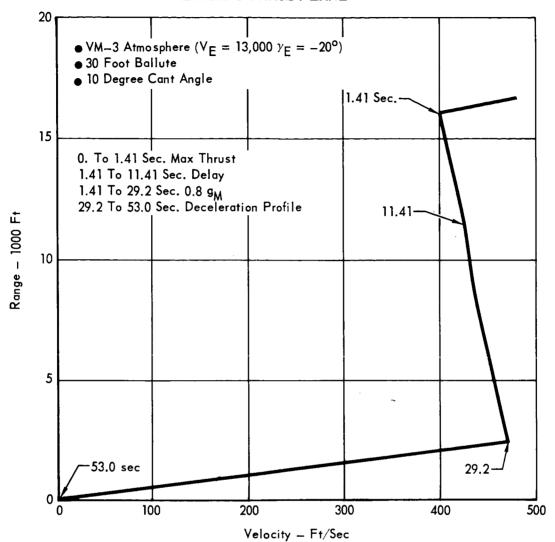
Initial conditions were obtained from computer entry runs with and without the Ballute. The following basic control sequence was employed:

- o Ballute was released prior to terminal propulsion ignition.
- o A maximum thrust phase was used for separation and was maintained to a preset velocity increment. The lander attitude was inertially held to the orientation at ignition.
- o After the separation thrust period, the thrust was throttled back to provide 0.8 g's. The attitude remained inertially fixed.
- o After a preset time, the attitude was aligned along the velocity vector.
- o Control to the deceleration profile was initiated when the measured range component of velocity equalled the pre-programmed velocity command. The programs examined lander control procedures involving a variety of altitude, separation thrusting times, lander cant angle and Ballute sizes. A 100 ft/ sec separation velocity increment at maximum thrust was selected for the following reasons:
- O Adequate separation clearance was provided with a 1.4 second maximum thrust period.
- o Even though smaller values could provide adequate clearance, propulsion considerations require that the burn time should not be shorter than 1.4 seconds.
- o Increasing the velocity increment had the effect of reducing the lander velocity such that it would not pass the Aeroshell for most of the various atmospheres.

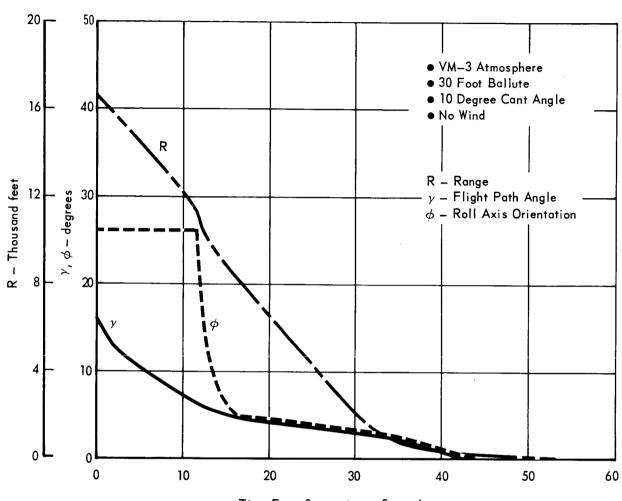
A separation altitude of 15.000 feet was selected to allow sufficient time for verticalization prior to switching to the deceleration profile in the high velocity cases. The cant angle of 10 degrees (lander to Aeroshell roll axes) provides adequate clearance even for the steepest trajectories and is consistent with possible design considerations. Another method having equally good results is to torque the lander as it separates from the Aeroshell and then torque back to the original alignment.

To illustrate the type of information supplied by these trajectory studies, Figure 4.6-3 shows a typical phase plane of a lander trajectory for a VM-3 atmospheric entry. Figure 4.6-4 provides lander range, flight path angle, and roll axis orientation time histories. The relative trajectories of the lander and the Aeroshell for both no wind disturbance and \pm 180 ft/sec steady winds are illustrated in Figure 4.6-5. Figure 4.6-6 shows the time history of the Aeroshell position relative to the axis system of the lander in order to answer questions concerning Aeroshell interference with landing radar acquisition. It was concluded

LANDING PHASE PLANE



LANDER TIME HISTORIES



Time From Separation - Seconds

Figure 4.6-5 4-144

RELATIVE POSITION OF AEROSHELL REFERRED TO LANDER AXES

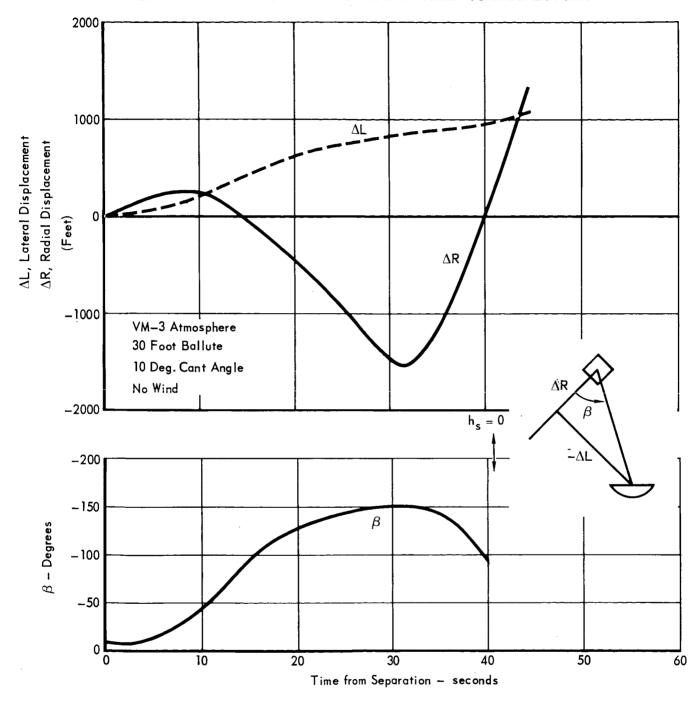


Figure 4.6-6 4-145

from these investigations that with the proper control configuration relative trajectories of the Aeroshell and lander could be maintained in all atmospheres with sufficient clearance for safe landing radar operation for the Fire-in-the-Hole separation concept.

For the Differential Drag separation concept, similar trajectory studies were undertaken to determine if the Aeroshell interfered with the landing radar operation after separation. Uncontrolled relative trajectories which were studied indicated that the Aeroshell would always be in front of the lander, a condition which would cause concern about radar interference. Therefore, further controlled trajectory studies were made with parachute sizing, deployment altitude, and Aeroshell release point designed to achieve Aeroshell impact on the Martian surface before there would be a need for landing radar information. A set of relative trajectories for the bounding VM-7 and VM-10 atmospheres which meet this requirement at lander altitudes of 4300 and 6700 ft respectively are presented in Section 2.3.7. A deployment altitude of 23,000 feet with Aeroshell release (separation) following 12 seconds later and using a parachute size of 70 feet was determined to be required for these trajectories. Higher deployment and release altitudes were found to be undesirable for the VM-10 atmosphere since the equilibrium velocity would be too low and, in the presence of continuous design winds, would result in too shallow a flight path angle. This angle would not be within the design constraints for radar lock-on when the roll axis is controlled to the velocity vector during the gravity turn control. Parachute release would follow immediatley unless there was a positive indication that terminal engine ignition had not taken place. Ignition would be programmed for 5000 feet altitude for all atmospheres. The trajectory and design data which was developed during this study indicated that terminal descent for this concept of separation, can be programmed so that there is no interference with the landing radar. For a more detailed discussion of this separation concept refer to Section 2.3.7

4.6.4.2 <u>Contamination Study</u> - The potential effects of exhaust plume contamination caused by Fire-in-the-Hole separation on experiment sensors and engineering transducers was a cause for some concern. In order to define these effects an effort was made to determine the level of contamination that might be expected. A "near field" analysis of exhaust contamination on the Aeroshell during the "pop-gun" separation interval was performed. While this investigation was limited to stagnation point contamination levels and could not be used quantitatively to assess exhaust effects on sensitive surfaces, it did reflect that there would be a very

substantial level of contamination present. The centerline contamination to be expected would be as follows:

Centerline Contamination

Engine Concept	gm/cm ²
Single	0.63
3	0.49
6	0.40

An examination of the various required sensors and transducers indicates that the expected contamination would primarily affect the entry TV. This would make it mandatory that the TV lenses be covered and sequenced for interruption during the separation thrust interval. The balance of the sensors and transducers do not appear to be affected seriously.

4.6.4.3 Thermal Investigation - Another potential problem associated with Fire-inthe-Hole separation was possible excessive heating at the base of the lander due to exhaust plume recirculation. Gas temperatures at the base region of the lander were expected to attain 3500 R over a .18 second interval. Using this as an initial condition, the temperature rise expected in the base region of the lander was determined. Sufficient heat capacitance was available to limit the expected temperature rise to only 70 F. This rise is not expected to constitute a problem. 4.6.4.4 Design Studies - A design study was undertaken to determine the scope of the problem of providing ports in the Aeroshell for the Fire-Through-Hole separation. The concept was found to be extremely sensitive to the lander configuration. The ducts for the rocket engine exhaust to get through the Aeroshell would be excessively long. In addition, the ports in the Aeroshell would cut directly through the nose cap. To avoid the nose cap and radar altimeter antenna requires "bending" the exhaust ducts, which would be an undesirable sacrifice in thrust efficiency. The size of the ports involved would be excessive in any case. Interfacing the engine nozzles on the lander with the exhaust ducts on the Aeroshell would also be difficult. In general, the design problems involved in using the Fire-Through-Hole separation concept imposes severe constraints on the design of the lander and/or Aeroshell.

A design study was also undertaken to determine feasibility of the "Split Aeroshell" concept. It was determined that the Aeroshell could be effectively split into three segments hinged to allow the differential drag device to provide part of the opening force. Additional opening force was provided by a favorable aerodynamic moment about the hinge points. The nose cap required could be attached to and remain a part of one of the three segments. This method of separation would

add complexity to the system, both in the structural detail and in the necessary additional pyrotechnics.

4.6.4.5 <u>Weight Study</u> - A prime factor in evaluating the separation techniques was the weight penalty incurred. The following table summarizes the system weights directly affected by the separation technique:

<u>System</u>	Differ- ential (Para- chute)	Differ- ential Drag Ballute	Segmented Aeroshell	Fire- in-the- Hole	Fire- Through- Hole
Parachute	195	-	_	-	_
Ballute	-	19 5	170	175	_
Terminal Propulsion	680	680	.805	905	1205
Blast Covers, etc.	-	· -	-	50	50
Aeroshell Segmentation	-	-	110	_	-
Hole in Aeroshell TOTALS Wt. (difference from lightest system)	875 lbs	8751bs	- 1085 lbs +210	- 11301bs +255	65 13201bs +445

The following values were used to derive these weights:

<u>System</u>	*Differ- ential (Para- chute) (VM-8)	Differ- ential Drag (Ballute)	Segmented Aeroshell		Fire- hrough Hole
Parachute Diameter	70 ft.	-	-	-	_
Ballute Diameter	-	47 ft.	43 ft.	30 ft.	-
Deployment Altitude	23,000 ft.	20,000 ft.	20,000 ft.	30,000 ft.	_
Separation Altitude	17,700 ft.	Approx. 19,000 ft.	10,000 ft.	15,000 ft.	Near Surface
Mach No. at Deploy.	2.0	2.3	2.3	4.1	-
Velocity at Separation	1350 fps	Greater than 600 fps	600 fps	800 fps	Low
Term. Prop. Tot. Impulse	65,000 1b-sec	65,000 1b-sec	90,000 1b-sec	110,000 1b-sec	170,000 1b-sec

^{*} Data added at end of study.

Deployment conditions resulting in the lightest Ballute weight consistent with the terminal propulsion initiation constraints were chosen for all techniques employing a Ballute. A lower limit of 20,000 feet was arbitrarily imposed on the Ballute-deployment altitude. While these systems were not fully optimized in this study, it appears unlikely that further optimization would change the order of weight preference. For Flight Capsule weights of 5000 lb. or less, changing from a tucked-back Ballute to a parachute system would not change the results of this study.

4.6.5 <u>Evaluation</u> - The four high value candidate Aeroshell separation concepts were evaluated using the method described in Section 1. The primary items which had a major influence on the ratings are summarized in Figure 4.6-7. The bottom of each box in Figure 4.6-7 also contains the relative score for that criterion. All four candidates were scored the same for cost since no major differences could be accurately identified. Weighted scores for the candidate concepts are given at the bottom of each column.

A more detailed discussion of the evaluation for each concept follows.

4.6.5.1 <u>Fire-in-the-Hole</u> - The contamination studies indicated that a serious contamination problem, primarily for the entry TV optics, did exist. The optics would have to be covered during separation and sequenced to open after separation was completed. There would therefore be a loss of TV data during separation and some degradation in subsystem reliability. In addition, the plume effects on other Capsule Bus, Surface Laboratory System, and Entry Science Package components are unknown and largely unpredictable at this time.

It appears that the problem of base heating of the lander because of high plume temperature is not serious. The expected temperature rise is calculated to be only 70 F, due to the extremely short period of high temperature exposure and the presence of sufficient thermal capacitance in the lander.

The trajectory studies have established with some degree of confidence that, with proper programming, the radar and physical interference problems can be avoided. While some element of unpredictability in the Aeroshell dynamics after separation exists, it appears that this will not affect landing radar operation. It is somewhat less certain that it will not cause some physical recontact.

The system weight penalty of 255 1b is the second lightest of the four and a very significant factor in the evaluation.

Since the application of the Fire-in-the-Hole technique to large bodies such as the Aeroshell presents dynamics problems of largely unknown scope, extensive

COMPARISON OF HIGH VALUE CONCEPTS - AEROSHELL SEPARATION

:	FIRE-IN-HOLE	FIRE.THROUGH-HOLE	DIFFERENTIAL DRAG	SPLIT AEROSHELL-DIFFERENTIAL DRAG REMOVAL
Probability of Mission Success Weight — .35	Contamination problem exists for sensors, primarily optics. Will require covers and sequencing thus reducing system reliability. Plume effects on other CBS, SLS and ESP elements uncertain at this time. No apparent thermal problem. Radar and physical interference problems appear avoidable.	No radar or physical recontact problem. Drag effects of firing into free stream almost totally unknown.	Relatively simple system, high reliability Radar and physical interference problems appear avoidable.	Simple reliable system, but less reliable than straight differential drag separation. No radar or physical interference problems since Aeroshell is always above Lander. Loss of TCM lock will occur at separation. Period of loss unknown but will vary with atmosphere. If failure occurs in separation and lock is not re-established no information on failure will be transmitted.
System Performance Weight — .2	255 lb. weight penalty Some uncertainty in Aeroshell dynamics after separation. (.6)	445 lb. weight penalty Degradation in velocity measuring accuracy of radar due to 'noise' of reflections from vibrating radome. Extreme difficulty in porting the Aeroshell for efficient rocket exhaust. (.2)	Baseline weight Good predictable per- formance. Best accumulation of subsonic entry science data. (.9)	210 lb. weight penalty Good predictable performance — no trajec. tory uncertainties.
Development Risk Weight2	Extensive wind tunnel testing necessary.	Radome for these temperatures never made in this size before. Extensive wind-tunnel testing necessary. Considerable radarradome compatibility testing required. Radome may provide severe constraint on antenna design.	Much parachute data available. Requires some flight test and wind tunnel testing, but probably less than the others.	Much parachute data available. Requires extensive flight test and wind tunnel testing.
Versatility Weight – .15	Good growth potential Not affected by variations in environment (.9)	Growth potential unknown Method sensitive to Lander geometry. Not affected by variations in environment. (.4)	Fair growth potential Not affected by variations in environment. (.8)	Excellent growth potential Not affected by variations in environment. (.9)
Cost Weight 1 Weighted Score	(.5)	(.5)	(.83)	5)

Figure 4.6-7

wind tunnel testing may be necessary. In general, the aerodynamic problems associated with any technique employing a propulsive Aeroshell/lander separation system will be more complex than one employing an aerodynamic decelerator.

The Fire-in-the-Hole concept has good growth potential since the available maximum thrust will always be considerably higher than the minimum required for separation. This technique is not particularly affected by variations in environment since it does not depend on an atmosphere for separation.

4.6.5.2 <u>Fire-Through-Hole</u> - No radar or physical recontact problem is predicted for this technique since separation would be delayed until a very low altitude (50 to 100 ft.) at which point the velocity of the Aeroshell/lander would have been reduced almost to zero.

A largely unknown factor in this concept is the effect on Aeroshell drag of exhausting the rockets into the free stream in front of it. However, available literature shows that drastic changes will occur in the shock wave shape, shock standoff distance, and to the Aerodynamic characteristics of the Aeroshell. These effects are largely dependent upon the number of jets, their position on the Aeroshell, and the ratio of jet total pressure to free stream total pressure. Best available predictions indicate that the Aeroshell drag will be drastically reduced. Additionally, the Aeroshell could lose its stability, becoming neutrally stable or even unstable. At the very least, an extensive wind-tunned and full scale testing program would be required.

The Fire-Through-Hole separation concept had the highest estimated weight penalty of the four concepts. This was, to a large degree, a function of the very high total impulse required. (170,000 lb-sec).

Some problems exist in connection with the use of the landing radar for this separation concept. Since this radar must operate "through" the Aeroshell, a radome would be required. No radome for the entry temperatures expected has ever been made in the size required. A degradation in the velocity measuring accuracy can be expected because of the "noise" caused by reflections from the vibrating radome. Typically, 1.0 ft/sec errors in horizontal velocity will be increased to 1.8 ft/sec. at altitudes of 100 feet. Considerable radar - radome compatibility testing is predictable. This would include tests to determine the vibration/acceleration spectrum for the radome, tests using the derived spectrum, and analytical and simulation work to determine the effects of radome reflections on radar performance for different trajectories.

One of the major design difficulties entailed in this separation concept is the problem of porting the Aeroshell efficiently for rocket exhaust. The interfacing of the exhaust ducts with the engine nozzles on the lander, the location of the ports relative to critical structure and components, and the large ports required in the Aeroshell are difficult design problems to overcome. These design problems also have an indirect detrimental effect on the versatility of the method since it is very sensitive to the configuration of the lander.

This concept is not sensitive to differences in environment since it is essentially all-propulsive.

4.6.5.3 <u>Differential Drag</u> - This candidate separation concept has the overriding advantage of being the simplest of the four high value candidates and therefore the most reliable, with a high probability of mission success. From an aerodynamic standpoint, it is easily the most predictable. It is therefore the simplest to work with from a design standpoint.

Trajectory studies indicate that the landing radar and physical interference problems can be readily avoided by tailoring the parachute size, deployment altitude, and Aeroshell release point to accomplish Aeroshell touchdown on the Martian surface before landing radar information is required.

While this concept will undoubtedly require some flight test and wind-tunnel testing, it should be substantially less than required for the others since it is dependent upon known parachute technology.

The Differential Drag separation method has reasonable growth potential, being limited only by the limits of parachute sizes and the attendant packaging volume. The feasibility of this method is not affected by variations in the environment since a difference in drag is present in any atmosphere. However, the rate of separation would be a function of the atmospheric density.

4.6.5.4 <u>Split Aeroshell - Differential Drag Removal</u> - This separation method is reasonably simple and therefore reliable; it is not as simple as the Differential Drag method. Its reliability is compromised somewhat by the requirement for a fairly extensive pyrotechnic or mechanical arrangement to effectuate the segmentation of the Aeroshell.

No landing radar interference problem exists with this method since, following separation, the Aeroshell is always above the lander. Since the aerodynamic forces on the Aeroshell - parachute combination will be vastly different from the forces acting on the lander, the chances of physical recontact are almost non-existent.

This separation method has one problem which the others do not. Immediately

following separation, a loss of telecommunications lock will occur because the Aeroshell will be between the lander and the Spacecraft. The length of this period of loss of lock has not been established but will vary with the time the lander's VHF transmitter is in the Aeroshell/Spacecraft line of sight. The information normally transmitted during this period will be stored and transmitted as soon as lock is again established. However, if a failure should occur during separation and lock is not re-established, no information on the failure will be transmitted.

This method of separation has an estimated 210 lb. weight penalty which makes it a poor second best concept.

This method has a relatively predictable performance regime since there are no trajectory uncertainties (the Aeroshell is always above the lander) and the actual mechanics of the separation process are relatively simple and straightforward.

The development risk entailed in this method is considered second to that of the Differential Drag method. While the parachute removal of the Aeroshell is simple, the opening of the Aeroshell and the release and clearance of the lander will require considerable flight and wind tunnel testing.

The Split Aeroshell technique has excellent growth potential. Only growth in Aeroshell weight will be reflected in parachute size and the method is not particularly sensitive to lander configuration. As with the Differential Drag method it is affected by variations in atmosphere only in the rate of separation.

4.6.6 <u>Recommended Design Approach</u> - As a result of the various interdisciplinary evaluations made during this trade study, the Differential Drag method of separation was selected as the approach having the highest merit. Parachutes are considered to be the best auxiliary aerodynamic decelerator for separation purposes.

- 4.7 ORBITAL DESCENT ATTITUDE REQUIREMENTS The purpose of this study is to define the attitude requirements imposed on the Flight Capsule during orbital descent by the telecommunications and thermal control subsystems, and to determine the consequent requirements for time of Sterilization Canister separation and the limitations of landing site availability.
- 4.7.1 <u>Summary</u> Separation of the Sterilization Canister, with the multilayer insulation blanket attached, just prior to spacecraft-capsule separation in Mars orbit is the preferred timing.

Utilization of a slow roll - 3 to 4 rev/hr - during the exoatmospheric descent, with alignment of the Flight Capsule axis to the entry attitude, provides continuous capsule-to-spacecraft communications for landing sites up to about noon, Martian time. Landings closer to the evening terminator require either the addition of an insulation blanket to maintain acceptable heat shield temperatures, or the qualification of a -300° F ablator.

- 4.7.2 <u>Requirements</u> The primary functional requirement imposed on this study is that the Sterilization Canister perform its primary function of preventing contamination of Mars by a contaminated Capsule Lander without violating the technical requirements of the Capsule Bus subsystems. These requirements are that capsule-to-spacecraft communications be continuous throughout descent from orbit, that the landing site be within 15° to 30° of the terminator, and that the ablative heat shield be kept warrier than -150°F.
- 4.7.3 <u>Design Approaches and Characteristics</u> The two operational characteristics requiring definition by this study were the time of Sterilization Canister separation and the thermal control technique during orbital descent.
- 4.7.3.1 <u>Canister Separation Timing</u> Five points of Sterilization Canister separation were considered, their most important characteristics being summarized in Figure 4.7-1. Examination of the features noted shows that separation in Mars orbit has several significant advantages if it can be shown that the canister would not decay from orbit prior to 1984. The most dense upper atmosphere model results from an extension of the VM-3 model, as given by Figure 4.7-2. Using this model and the analytical method of Section 2.3.1, we obtain the relationships shown in Figure 4.7-3. The canister preferred design has a ballistic parameter (M/C_DA) of about 0.02 slugs/ft for a randomly oriented body in free molecular flow. Thus, to provide a ten-year orbital lifetime requires that the periapse altitude be at least 720 km. This requires a deviation from the General Speci-

STERILIZATION CANISTER SEPARATION TIMING ALTERNATIVES

(1) At Interplanetary Insertion	 No contamination of Mars due to separation debris No micrometeoriod protection from canister Lowest canister weight, but increased Aeroshell weight to provide equivalent micrometeoroid protection Requires insulated heat shield 	
(2) Just Before Mars Orbit Insertion	 Low probability of contamination Requires insulated heat shield; effect of micrometeoroid impacts on insulation uncertain Lowest weight to be orbited; thus, less restricted operations window due to fixed S/C propulsion 	
(3) In Mars Orbit, Just Prior to S/C — CBS Separation	 Possibility of orbital decay of canister or debris causing contamination Canister provides micrometeoroid protection in orbit Allows placement of insulation blanket on canister rather than heat shield, eliminating a separation sequence Allows greater flexibility in selection of thermal control technique 	
(4) After S/C — CBS Separation	• Same as (3), but more complex separation; heavier	
(5) None	 Canister hinged, remains attached to S/C Most complex; heaviest 	

VM-3 MODEL ATMOSPHERE EXTENSION

ALTITUDE	DENSITY IN SLUGS/FT ³	
(Km)	TIME: 0400	1400
103	3.98 × 10 ⁻⁸	3.98 × 10 ⁻⁸
157	1.37 × 10 ⁻⁹	1.99 × 10 ⁻⁹
212	8.74 × 10 ⁻¹¹	2.05×10^{-10}
270	8.60 × 10 ⁻¹² \	3.28 × 10 ⁻¹¹
329	1.13	7.14 × 10 ⁻¹²
390	1.88 × 10 ⁻¹³	1.92
453	3.79×10^{-14}	6.04 × 10 ⁻¹³
518	9.02 × 10 ⁻¹⁵	2.17
586	2.39	8.57 × 10 ⁻¹⁴
655	6.98 × 10 ⁻¹⁶	3.68
731	2.25	1.70
808	_	8.24 × 10 ⁻¹⁵
823	6.48 × 10 ⁻¹⁷	-
887	3.09	4.21
923	_	3.16
1054	4.85 × 10 ⁻¹⁸	1.20
1234	7.61 × 10 ⁻¹⁹	3.47 × 10 ⁻¹⁶
1431	1.20	1.00
1643	1.88 × 10 ⁻²⁰	2.89 × 10 ⁻¹⁷
1877	2.95 × 10 ⁻²¹	8.34 × 10 ⁻¹⁸
2135	4.62 × 10 ⁻²²	2.41
2417	-	6.96 × 10 ⁻¹⁹
2731	1.00×10^{-22}	2.01

From:

D.N. Vachon, "On the Distribution of Density at Orbital Altitudes in the Martian Atmosphere," General Electric Co. Space Physics Technical Memorandum No. 8126—5, 30 June 1966.

ORBITAL SIZE TO PROVIDE 10 YEAR ORBIT LIFE EXTENDED VM-3 ATMOSPHERE

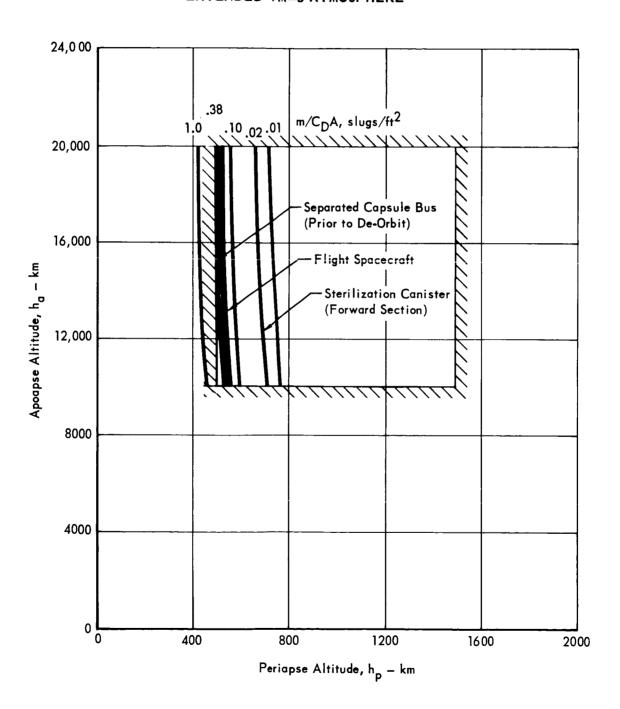


Figure 4.7-3 4-157

fication that periapse altitudes of 500 km be acceptable. However, the atmosphere extension assumption was a conservative one, and it is possible that Mariner 1969 will indicate a less dense upper atmosphere, which would permit operation at lower orbital altitudes.

If the Sterilization Canister is separated prior to Mars orbit insertion, an insulation blanket must be installed over the ablative heat shield. This blanket must be separated before aerodynamic heating commences to preclude melting, thereby potentially degrading the performance of heat-shield mounted sensors. Canister separations prior to Mars orbit insertion also results in potential contamination of the capsule by the non-sterile exhaust of the space-craft propulsion subsystems during orbit insertion and attitude control maneuvers. Retention of the canister during orbital flight of the capsule prevents such contamination.

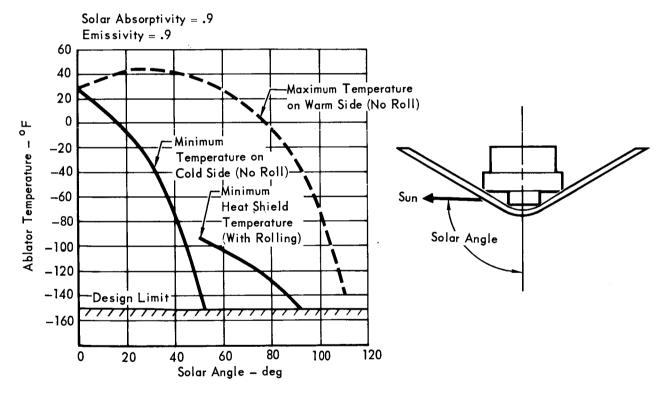
4.7.3.2 <u>Thermal Control</u> - The basic thermal control problem of interest to this study is to maintain heat shield temperatures at acceptable levels. The choice of techniques is dependent in part on the time of canister separation.

Exposure of the heat shield to space without benefit of solar heating causes the ablator to cool rapidly to about -300° F, after which it stabilizes. While the preferred ablator has satisfactory performance at that temperature, we would prefer to operate at somewhat warmer temperatures to provide added margins. Figure 4.7 -4 shows the equilibrium temperature of the ablator as a function of the solar angle, defined as the angle between the roll axis of the Flight Capsule and the Capsule-Sun line, for the 120-degree sphere-cone configuration. Choosing -150° F as a desirable lower operating temperature, we see that a solar angle of 50 degrees is required. Provision of a slow roll - about 3 to 4 rev/hr - increases the allowable solar angle to 90 degrees.

Figure 4.7-5 summarizes the consequences of the candidate thermal control techniques. It is noted that the insulated ablator is the only approach (other than the -300° F ablator) capable of surviving a canister separation prior to Mars orbit insertion, while all the techniques are applicable to an in-orbit separation.

Canister separation in orbit permits the insulation blanket to be installed on the canister, rather than on the heat shield. The primary advantage is the elimination of a separation sequence. Figure 4.7-6 shows the comparative features of insulation location, internal or external to the canister. The external location is strongly preferred.

ABLATOR TEMPERATURES AT END OF ORBITAL DESCENT



SUMMARY OF THERMAL CONTROL ALTERNATIVES

	DE-ORBIT			CANISTER SEPA	RATION CAPABILITY
HEAT SHIELD CONFIGURATION	SOLAR ANGLE REQUIREMENTS	WEIGHT PENALTY	LANDING SITE CONSTRAINTS	BEFORE ORBIT	BEFORE CAPSULE SEPARATION
Bare Ablator (-150°F)	0 – 50°	0	Early Morning <25 ° from Morning Terminator		V
Bare Ablator With Slow Roll	0 – 90°	- 0	Before Noon	,	, √
Insulated Ablator	0 – 180°	0	None	•	▼
Gold Tape Over Ablato	r 0 – 180°	26	None	Marginal	1 √,
Low Temp. Ablator (-300°F)	0 – 180°	< 12 (Added Heater Power)	None	y	₩

COMPARISON OF ALTERNATIVE INSULATION BLANKET LOCATIONS

INSULATION LOCATION	ADVANTAGES	DISADVANTAGES
Inside canister	Insulation not subjected to micrometeoroid or thruster damage.	 Canister exposed to very low temperatures (-350°F). Insulation must survive terminal sterilization cycle. Canister cannot be pressurized after launch. Complex canister separation.
Outside canister	 Would result in canister material and separation devices being at a temperature of approximately -150°F. Multi-layer insulation would not have to be exposed to terminal sterilization cycle. Simpler installation - better support for launch and separation inertia loads. Standardization enhanced for routing and checkout equipment. Simplified canister ejection. Possible micrometeoroid protection from insulation blanket. 	Multi-layer insulation could possibly be damaged by micrometeoroid impingement and spacecraft attitude control rocket exhaust.

4.7.4 Evaluation - A preliminary evaluation of the canister separation time alternatives shown in Figure 4.7-1 reduced to two the candidates of major interest: just before Mars orbit insertion or just before capsule-spacecraft separation in orbit. The evaluation of the applicable descent thermal control techniques for these two canister separation times, is summarized in Figure 4.7-7. Note that only four of the standard rating criteria are used; cost was not included since all are approximately equal. The remaining four criteria were further divided into several factors and then weightings were applied. In all cases, one of the candidates was assigned a maximum score and the others evaluated relative to it.

Based on this evaluation, canister separation in Mars orbit, using the slow roll for thermal control, is the preferred approach. The bare ablator is penalized by the severely restricted landing site flexibility. It is also sensitive to orbit inclination and the calendar date of de-orbit, thereby interacting adversely with the Flight Spacecraft system. These more than compensate for the increased reliability of the completely passive approach.

The insulated ablator shows up best in the pre-orbit case, due to the decreased probability of violating planetary quarantine (no decay of the canister from orbit). In both separation timing cases it benefits from a greater flexibility to changing environment and mission requirements, and provides an unrestricted landing site capability. However, the reduced reliability due to the required separation of the insulation blanket, the possible interaction with the heat shield and sensors, and the unknown effect of micrometeoroid damage to the insulation blanket outweight the advantages.

The cold ablator suffers primarily in the development risk associated with qualification to the $-300^{\circ}F$ temperature and the low confidence attendent to the reduced operating margins.

The selection of canister separation timing is predicated on either the acceptability of orbital altitude of 720 km of greater at periapse, or on the probability that the Mariner 1969 results will provide confidence that lower orbital altitudes would not cause orbital decay of the separated canister in less than the specified 10 years. If it does not, and the orbital decay represents an unacceptable risk at the time that a final design decision must be made, a change would be implemented to separate prior to Mars orbit insertion, which, as shown earlier, entails the installation of an insulation blanket over the ablative heat shield. The reliability penalty imposed by the requirement to separate this blanket and the uncertainty of the blanket susceptability to micrometeoroid damage is considered

CANDIDATE EVALUATION SUMMARY

		.S	TERILIZ	STERILIZATION CANISTER SEPARATION TIME	IISTER SEP	ARATION T	IME
			Z	IN ORBIT		PRE-	PRE-ORBIT
		BARE ABLATOR	SLOW ROLL	SLOW INS COLD ROLL ABLATOR ABLATOR	COLD ABLATOR	INS ABLATOR	COLD ABLATOR
Probability of Mission Success Reliability Effect in Other Systems	(.35) .15 .08	30. 91.	£1. 89.	60°.	90. 90.	.09 .05	99.5
Performance Weight Landing Site Flexibility	.10	5. 0. 9.	.10	2. 01.	.09	.10	. 00. 01.
Development Risk • State-of-Art Improvement • Confidence	.09	.09 11.	.09	.07	.05	.07	.05 .06
Flexibility • Environmental • Requirements Total	.08	.05	8.8.	.07	.00	.08	.0. .0. .0.
Loigi		٥,٠)	>	?	 ;) :

Figure 4.7-7

less desirable than the loss of low altitude orbit capability. It must also be remembered that if the Flight Spacecraft requirements place a high value on the very low orbit altitudes, the Flight Spacecraft orbit can be trimmed after the Flight Capsule is separated.

The selected thermal control mode requires solar heating to maintain heat shield temperatures above the design limit of -150° F. This restricts the capsule to morning landings. If, at some future date, it is determined that a landing near the evening terminator is required to improve the Flight Spacecraft performance, two alternatives are available:

- (1) The insulation blanket could be moved from the outside of the canister to the heat shield, as discussed earlier.
- (2) The development program necessary to qualify the ablator to -300° F operation would be initiated.

- 4.8 INDEPENDENT DATA PACKAGE STUDY The purpose of an Independent Data Package (IDP) is to add functional redundancy to the VOYAGER and thereby improve the probability of obtaining some diagnostic and surface environmental data. This study was conducted to examine the utility of an IDP as an adjunct to the entry and landing portion of the first VOYAGER mission. The parameters included in this study were reliability, weight, development risk, state of the art limitations, IDP integration into the Flight Capsule, and effectiveness.
- 4.8.1 <u>Summary</u> The usefulness of an IDP was evaluated by adding, as improvements, an equivalent weight to the Flight Capsule. The Flight Capsule with an IDP and the Flight Capsule with improvements were compared using reliability and system effectiveness analysis. In addition, the IDP development problems and its installation into the Flight Capsule were investigated. As a result of this study, the IDP was not incorporated into our preferred design.
- 4.8.2 Requirements and Criteria Special constraints are established in references 4.8-1 and 4.8-2. The constraints which affect this study are given in paragraph 4.1.2.3.1 of the constraints document (Reference 4.8-1) and paragraph 3.1.3 of the General Specification (Reference 4.8-2).

The IDP concept was evolved and investigated in accordance with these paragraphs. Achievement of a Flight Capsule landing, performance of entry science experiments, and performance of landed science experiments are the competing characteristics considered in our analysis of the IDP concept. Note that performance of landed science experiments is fifth priority and also the lowest priority of the competing characteristics which applied to the IDP.

- 4.8.3 <u>Design Considerations</u> The limitations imposed by the IDP design were evaluated to determine the overall impact on the Flight Capsule. This evaluation is presented in the following subsections.
- 4.8.3.1 Preferred Design Description The IDP subsystem would monitor critical Capsule Bus and Surface Laboratory engineering data; separate from the Capsule Bus or lander early in the descent sequence; descend to the surface via parachute; survive omni-directional impact at 250 ft/sec; and transmit the engineering and surface science data direct to Earth. The general characteristics of the subsystem and basic science instrument complement are given in Figure 4.8-1. The design constraints, optimization studies, and supporting analyses which were conducted to establish this configuration are presented in Section 5.15. The preferred concept employs a separable, hard landing, disk shaped capsule which is deployed near Aeroshell/lander separation. The essential elements of the landed payload and the

INDEPENDENT DATA PACKAGE GENERAL CHARACTERISTICS

BASIC CONFIGURATION

- Disk: 38 Inches Diameter x 14 Inches High
- Omnidirectional Impact Protection
- 250 Ft./Sec. Design Impact Velocity
- 3100 g Peak Impact Deceleration
- Parachute Descent Retardation
- 100 Pounds Gross System Weight
- Payload Size; 15.6 Inches Diameter x 5 Inches High
- Payload Weight Fraction 0.5 (Nominal)
- Balsa Wood Impact Limiter (6 lb/ft³)
- Two Atmospheric Sensor Masts (Selective Deployment)
- Six Fixed Cavity-Backed Cross Slot Antennas
- 4π Steradian Data Transmission
- 24 Hour Surface Operating Lifetime
- Silver-Zinc, 25 Watt-Hour/Pound, Battery
- Direct MFSK Telecommunication Link
- 20 Watts Transmitter Output Power, 1.2 BPS
- 800 Bit Magnetic Core Memory

BASIC INSTRUMENTS

- Vibrating Diaphragm Pressure Transducer
- Gas Chromatograph for Atmospheric Composition
- Hygroscopic Sensor for Water Vapor Detection
- Hot-Wire Anemometer for Wind Velocity
- IDP/CB Diagnostic Sensors

IDP subsystem as they would appear installed on the Capsule Bus are shown in Figures 4.8-2 and -3. A simplified functional block diagram is shown in Figure 4.8-4.

- 4.8.3.2 Installation of IDP into Flight Capsule Installation of the IDP within the Flight Capsule must consider many factors, principally:
 - a Locating the IDP off the centerline requires additional weight for ballast and may have adverse affects on the reaction control subsystem when the IDP is deployed.
 - b Locating the IDP on the Surface Laboratory requires beefed-up support structure and may interfere with externally mounted experiments.
 - c The physical location should not interfere with or degrade the overall performance of the Flight Capsule.
 - d A simple, highly reliable deployment technique should be used.

The installations considered most desirable are shown in Figure 4.8-5. A summary of the major problems encountered while trying to install the IDP in our baseline design is given in Figure 4.8-6.

4.8.3.3 Deployment Techniques - The deployment of the IDP for all preferred separation altitudes requires pyrotechnic devices, parachute, sequencing and timing, and electrical power. The IDP deployment is a complicated procedure. Deployment must occur with minimum reaction torque on the Flight Capsule to prevent tumbling the Capsule Bus during the terminal descent phase. Three deployment sequences were considered for each IDP location:

Forward Location

Aft Location

- a Deployment through nose a Deployment prior to parachute deployment
- b Deployment from lander
- b Deployment from de-orbit motor structure
- c Deployment from Aeroshell
- c Deployment from lander
- A typical sequence is given below for each location:

Forward Location - Aeroshell Deployment - As the IDP is released with the Aeroshell at 18,000 feet, the following events occur:

- a IDP remains on the Aeroshell (Time Delay)
- b IDP released from Aeroshell Section (5000 feet) by exploding bolt holding clamp ring.
- c IDP parachute released at separation (5000 feet)
- IDP descent on parachute d
- IDP parachute separation (50 feet or below)
- IDP hard landing f

INDEPENDENT DATA PACKAGE PAYLOAD

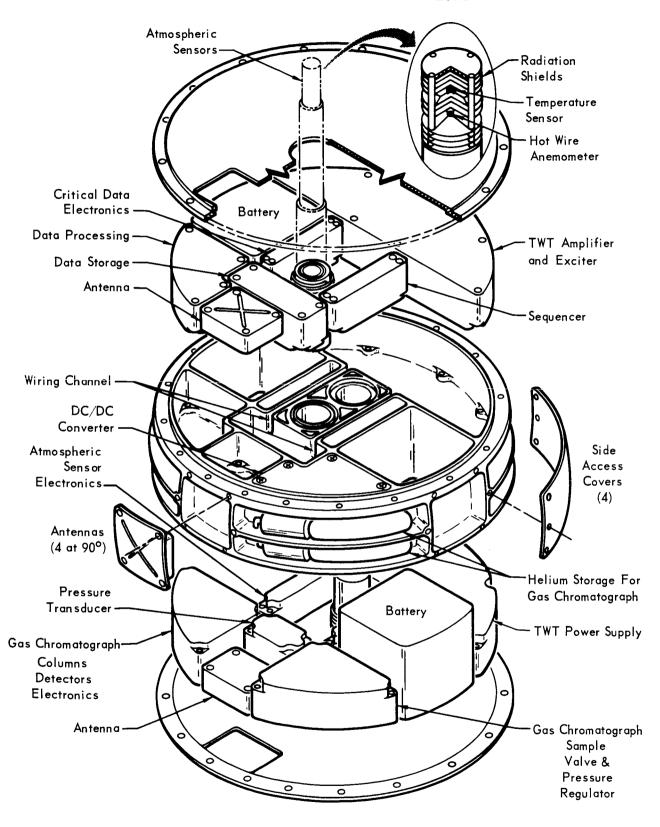
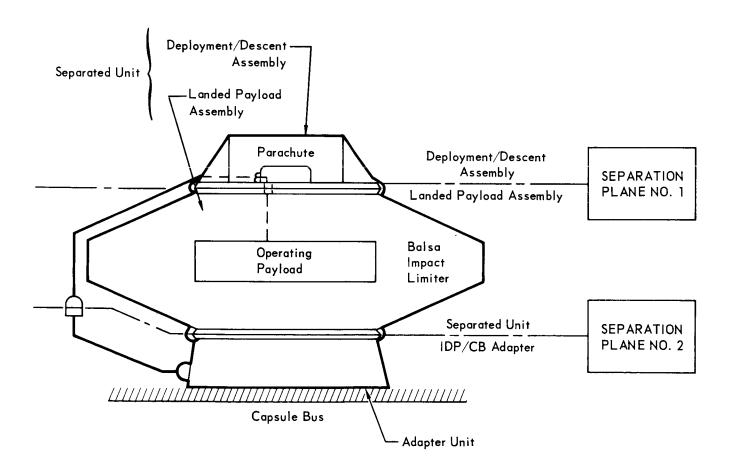


Figure 4.8-2

INDEPENDENT DATA PACKAGE SUBSYSTEM



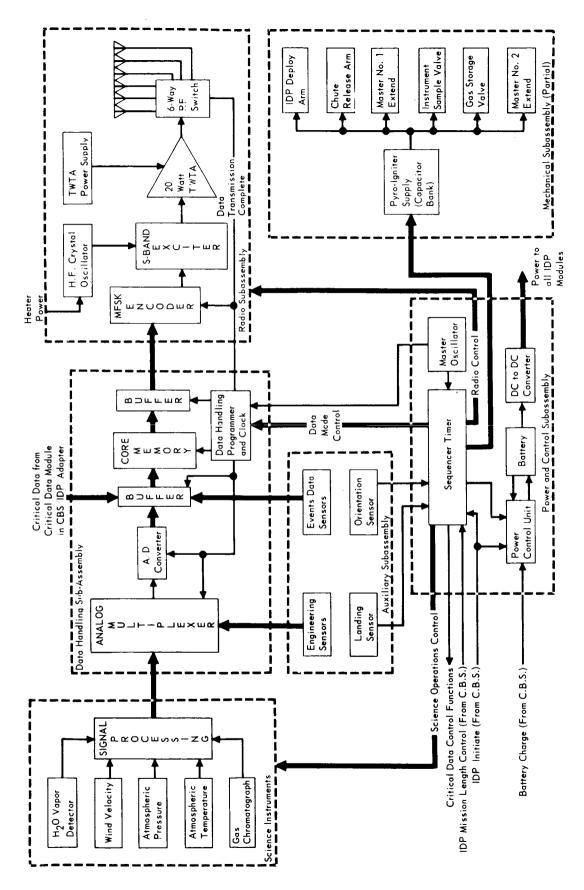


Figure 4.8-4 4-170

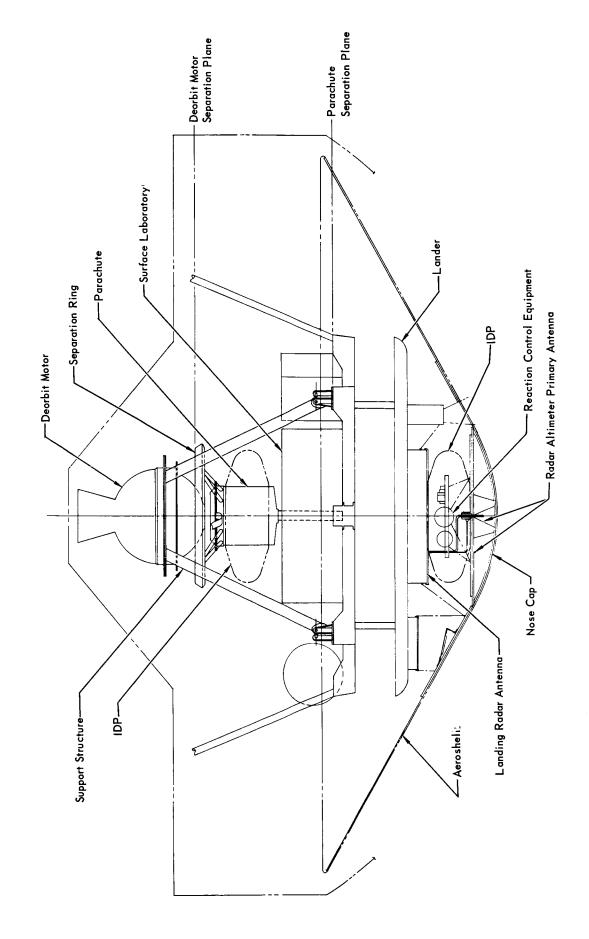


Figure 4.8-5

4-171

SUMMARY OF IDP INSTALLATION PROBLEMS

FORWARD AEROSHELL INSTALLATION

- o The IDP configuration does not fit in the space forward of the Lander without modification of the radar antenna structure.
- o The IDP is inaccessible after installation.
- o The reaction control equipment must be relocated.
- o If deployment fails, the IDP may not survive impact.
- o The IDP position causes major deployment sequencing problems:
 - 1) Deployment through the nose.
 - a) The present radar design cannot be used.
 - b) Design of the Aeroshell structure is complicated by the need for hinge and/or cutting mechanism.
 - 2) Deployment by removal with Lander.
 - a) Additional weight of the IDP, IDP attach structure, and IDP parachute system could tend to retard separation of the Lander from the Aeroshell
 - Failure of the IDP to deploy from the Lander would nullify the benefits of the Lander design.
 - Deployment from Aeroshell after Lander separation
 - a) Failure of release mechanism to operate would cause the IDP to remain entrapped in the Aeroshell.

AFT DE-ORBIT STRUCTURE INSTALLATION

- o The IDP position causes unsatisfactory relocation of the parachute off the center of gravity.
- o Mounting of external equipment on the Surface Laboratory is limited.
- o Additional structure is required for IDP mounting to the de-orbit motor support structure.
- o If the IDP fails its mounting, it would crush any externally mounted Surface Laboratory equipment.
- If deployment fails, the S-Band (high rate) antenna cannot be deployed and would severely limit T/M data transmission.
- o The IDP must be shielded from high entry heat.
- o If deployment fails, the IDP may not survive impact.
- o The IDP position causes major deployment sequencing problems.
 - 1) Deployment prior to parachute deployment.
 - a) Additional booster system is required to eject the IDP.
 - b) If deployment fails, the Surface Laboratory deployment experiments located beneath the IDP are unusable.
 - Deployment with main parachute separation from Lander. (IDP remains with de-orbit motor structure).
 - a) Additional weight of the IDP, IDP attach Structure, and IDP parachute system could tend to retard separation of (1) Lander from Aeroshell and (2) parachute and de-orbit motor structure from the Lander.
 - b) Low altitude parachute separation (less than 5000 feet) provides little time for deployment of the IDP parachute.
 - 3) Deployment with main parachute separation from Lander (IDP remains with Lander).
 - a) Additional weight of the IDP, IDP attachment structure, and IDP parachute subsystem could tend to retard separation of Lander from Aeroshell.
 - b) If deployment fails, the extra weight may degrade Lander landing performance.

Aft Location - De-orbit Motor Structure Deployment - IDP deployment from the de-orbit motor structure requires a two step structure separation rather than one step. This results from the need to keep the IDP mounting structure until IDP deployment.

- a De-orbit motor separation (800,000 feet)
- b Lander parachute deployment (23,000-20,000 feet)
- c Aeroshell separation (prior to 18,000 feet)
- d Lander on parachute (timed delay)
- e Parachute release (5,000 feet) and Terminal Propulsion System activates
- f IDP released from Capsule Lander
- g IDP separation releases parachute
- h IDP parachute separation (50 feet or below)
- i IDP hard landing

4.8.3.4 <u>Development Problems</u> - There is a degree of development risk associated with the IDP design. The cost to design, test and manufacture five IDP's could be \$23 to 30 million. Five vehicles (two IDP's for Flight, two for back-up, and one for testing) are required for the 1973 mission. The potential development problems are sterilizable high g batteries, impactable 20 watt transmitter, and impactable instruments.

Sterilizable High g Battery - A key problem area in the IDP design is the method of power generation. The best solution at this time is the silver-zinc battery. Several studies are in process to determine the best design for a silver-zinc battery to survive the two major environmental requirements of VOYAGER sterilization and a 3100 g impact. Although higher estimates have been given by some battery manufacturers, the best conservative estimate for battery specific energy is 25 Wh/lb.

Impactable Transmitter (20 Watt) - Hardware design problems include those of crystal oscillator instability and traveling wave tube amplifier design. The crystal oscillator design is especially difficult in the case of the IDP since a shock level of 3100 g is combined with the wide temperature variation during a Mars diurnal cycle. In order to withstand the shock, and to reduce the crystal oscillator drift as a function of changing temperature, it is necessary to house the crystal and the oscillator and buffer stages within a shock resistant isothermal environment. It is apparent that a traveling wave tube amplifier (TWTA) is necessary to generate efficiently 20 Watts of RF power at S-band. This approach presents a problem in the

case of the IDP shock environment (3100 g). Watkins-Johnson Inc. has done the only known work to-date in implementing a shock resistant TWTA. Their tube, Model No. WJ-398 (22 Watts at S-band), has been successfully tested at a 10,000 g peak, 1 millisecond duration shock level.

<u>Impactable Instruments</u> - The instruments considered for the IDP and their development status are given in Figure 4.8-7. The necessary instruments should be developed in approximately two years; however, the capability required is beyond the present state of the art.

4.8.3.5 <u>IDP Operation After Landing</u> - The uncertainties of the horizontal wind velocity and Martian surface will have a significant affect on the successful operation of the IDP. Throughout this study the assumption was made that the IDP would land and operate satisfactorily. However, the landing loads on the IDP and its final position could prevent instrument mast deployment and/or cause damage to some of the antennas. These and similar types of landing problems could limit or prevent useful data being transmitted by the IDP. In evaluating the IDP these uncertainties must also be considered along with all the other facts presented in this study.

4.8.4 Evaluation - After the IDP design was established, the value of the IDP as a part of the Flight Capsule was analyzed. The weight required for the IDP-100 1b-can be allocated in three alternate ways: (1) the IDP can be incorporated into the design; (2) the 100 1b can be used to improve the Surface Laboratory through redundancy additions; (3) the 100 1b can be used to improve the effectiveness of the Flight Capsule by the technique described in Section 4.10. The uncertainties of the IDP and Capsule Bus System interference (i.e. recontact, parachute entanglement, parachutes landing on IDP or Surface Laboratory) and the Martian surface conditions affects on the IDP landing were not included in this analysis.

4.8.4.1 Reliability Analysis - The reliability analysis was conducted based on obtaining minimum, low rate surface environmental data. For simplicity in conducting the analysis, the instrument reliability (for the surface pressure, temperature, wind speed, water vapor, and composition measurements) in the Surface Laboratory and IDP were assumed to be the same so they were not included in the IDP reliability estimates. Inclusion of the experiments would not significantly affect the results.

Eleven different configurations were initially analyzed for reliability, including eight different IDP release and deployment sequence times during the descent and landing mission phases. The IDP release and deployment sequence

IDP INSTRUMENT DEVELOPMENT STATUS

INSTRUMENT	LOCATION	REMAINING DEVELOPMENT
Vibrating Diaphragm Pressure Transducer	Interior (Electronics) — Access port for static pressure	Completion of integrated circuit design Shock hardening Sterilizability Production methods
Platinum Resistance Thermometer	Interior (Electronics) – Sensor deployed on extendable mast	Completion of integrated circuit design Completion of deployment method and radiation shield design Shock hardening Sterilizability
Hot Thermocouple Anemometer	Interior (Electronics) – Sensor deployed on expendable mast	Completion of integrated circuit design Completion of deployment design Completion of study of calibration methods Study of atmospheric composition effect Shock hardening Sterilizability
Gas Chromatograph	Interior	Fabrication and test of the two column gas chromotograph Completion of study of sampling and cali- bration methods Shock hardening Perfection of double dynamic range Sterilizability
Hygroscopic Water Vapor Sensor	Interior	Completion of integrated circuit design Temperature effects Sensitivity Calibration methods Shock hardening Sterilizability

analysis is presented in Section 5.15.3. Four configurations were selected for detailed analysis as follows:

- <u>Configuration 1</u> Baseline Flight Capsule without IDP.
- Configuration 2 Baseline Flight Capsule without IDP but with 100 lb. of reliability improvements incorporated into the Surface Laboratory only.
- <u>Configuration 3</u> Baseline Flight Capsule with IDP; IDP is released from the de-orbit motor structure or the Aeroshell, but prior to terminal propulsion motor ignition.
- Configuration 4 Baseline Flight Capsule without IDP but with 100 1b of reliability improvements incorporated anywhere within the Flight Capsule

The reliability estimates include values for an IDP, an individual Flight Capsule, an individual Flight Capsule with an IDP, dual Flight Capsules, and dual Flight Capsules each with an IDP. The estimates are based on partial mission success (minimum surface experiment data) and include all Flight Capsule mission phases beginning at launch. The estimates do not include unreliability associated with the Flight Spacecraft and Launch Vehicle.

Figure 4.8-8 summarizes the reliability estimates. Configurations 2, 3 and 4 were compared to the Baseline (Configuration 1) by use of the Reliability Improvement Factor (RIF). The RIF is the ratio of the natural logarithms of the estimated reliabilities. The RIF is a measure of the reduction in unreliability and, therefore, is an indicator of the reliability improvement. Compared to Configuration 1, the reliability improvement factor is 1.5, 2.5, and 5.82 for Configurations 2, 3, and 4. Configuration 4, therefore, is the best based on the numerical reliability estimates. In addition to the numerical estimates, failure modes and effects, including critical single point failure possibilities, were considered in this analysis. The reliability models of the four Configurations studied are shown in Figure 4.8-9.

Configuration 2 (Improved Surface Laboratory) - This configuration enhances the reliability of Surface Laboratory electrical power, sequencer and timer, and telemetry subsystems. No major function single point failures are totally by-passed. Capsule Bus terminal propulsion and landing radar reliability are unchanged from the baseline design.

Configuration 3 (Baseline with IDP) - This configuration is the only design which effectively by-passes four major function single point failure possibilities, namely,

INDEPENDENT DATA PACKAGE RELIABILITY ESTIMATE

·	EQUIPMENT RELIABILITY					* RELIABILITY
CONFIGURATION	FC	2 FC	IDP	FC + IDP	2 FC + 2 IDP	IMPROVEMENT FACTOR
1(Baseline w/o IDP)	.858	. <u>977</u>	_	_	_	1.00
2(Baseline + 100 lb in S.L.)	.882	. <u>984</u>	_	_	_	1.44
3 (Baseline + IDP)	.858	_	.841	.908	. <u>991</u>	2.50
4 (Baseline + 100 lb in FC)	.942	. <u>996</u>		-	_	5.82

* R.I.F. =
$$\frac{\text{Baseline (LnR)}}{\text{Configuration (LnR)}}$$

RELIABILITY MODE

MINIMUM SURFACE DA

IDP RELEASE PRIOR TO TERMINAL

FC Electrical — FC Staging — CB Sequencing — CB Guidance — CB Attitude Control and — De-Orbit Propulsion and Timing and Control Power

CONFIGURATION 2 (100 LB. OF IMPROVEMENTS ADDED TO THE SURFACE LABORATORY)

CB Attitude Control and FC Electrical — FC Staging — CB Sequencing — CB Guidance and Timing and Control De-Orbit Propulsion

CONFIGURATION 3 (BASELINE WITH IDP)

FC Electrical — FC Staging — CB Sequencing — CB Guidance and Control — CB Attitude Control and — Less the Landing Radar De-Orbit Propulsion and Timing

CONFIGURATION 4 (100 LB. OF IMPROVEMENTS ADDED THROUGHOUT THE FLIGHT CAPSULE)

FC Electrical — FC Staging — CB Sequencing — CB Guidance — CB Attitude Control and and Control De-Orbit Propulsion and Timing Power

 $\sqrt{-}$ Denotes the functions which have improvements.

Figure 4.8-9

4-178 -1

L - IDP STUDY

TA RETRIEVAL PROPULSION MOTOR IGNITION

P_s (FC) = .858

CB Terminal SL Sequencing SL Tele-
Propulsion and Timing communications
$$P_s$$
 (2 FC) = .977

The second contraction is a second contraction in the second contraction in the second contraction is
$$P_s$$
 (FC) = .882. The second contraction is P_s (2 FC) = .984.

The second communication is
$$\frac{\sqrt{}}{\sqrt{}}$$
 and Timing $\frac{\sqrt{}}{\sqrt{}}$ $\frac{\sqrt{}}{\sqrt{}}$ $\frac{P_s (FC) = .942}{\sqrt{}}$ $\frac{P_s (FC) = .942}{\sqrt{}}$

Capsule Bus terminal propulsion and landing radar, and Surface Laboratory sequencing and timing and telecommunications. The other design configurations totally bypass none. Capsule Bus terminal propulsion and landing radar are considered to be the most critical single point failure possibilities because proper performance of all four engines and the radar is required for landing survival and retrieval of minimum surface environmental data.

Configuration 4 (Improved Flight Capsule) - The reliability of all major functions is improved with the exception of Capsule Bus terminal propulsion. Reliability improvements for the terminal propulsion function requires too large a share of the 100 lb and still neither improves the numerical reliability estimate significantly nor alters the critical single point failure possibilities (4 of 4 engines) for this subsystem.

4.8.4.2 IDP Value Assessment - The IDP deployment point (Concept A) does minimize the chance of IDP interference with the Surface Laboratory. If both the Surface Laboratory and the IDP land and operate successfully, the separation distance will enhance the measurements through correlation of similar measurements, and by measuring data in an uncontaminated area. The relative value of the IDP and the Surface Laboratory data is very difficult to define. The nominal information capacity of the Surface Laboratory is 450,000 non-imaging bits compared to 800 bits for the IDP. The IDP data is clearly a small addition to the total number of bits. However, the IDP data, being obtained from an uncontaminated area, may make this small amount of data of significant importance. If the Surface Laboratory lands and operates nominally, the primary value of the IDP is to supply composition data from an uncontaminated area. For this case we estimate the overall value of the additional IDP data to be 17% of the total Surface Laboratory value. If the Surface Laboratory does not land successfully, the IDP data becomes more important, it being the sole source of surface data. Thus, we consider the overall value of the IDP data to be 33% of the total Surface Laboratory value in this case.

4.8.4.3 Effectiveness Analysis - The IDP contribution to the total system effectiveness was evaluated relative to the effectiveness of adding 100 lbs of improvements to the Flight Capsule. The comparison was made using the effectiveness technique described in Section 410. The study considered both the total system effectiveness (E) and the effectiveness of achieving landed experiments (E₃). However, when an IDP is added the effectiveness model is modified as shown in Figure 4.8-10. Then the term $E_3 = V_3 R_3$ must be redefined:

R₁ Achievement of a Flight Capsule Landing R₃ Performance of Landed Science Experiments R₂ Performance of Entry Science Experiments Landed Experiments THREE PHASE MISSION RELIABILITY/EFFECTIVENESS ANALYSIS Experiments Entry \in Ê CBS Electric Power CBS Telecom (1) Not included in the calculations Telecom Telecom Telecom ESP ESP SLS RELIABILITY DIAGRAM Guidance and Control Sequencer SLS RCS and De-orbit Propulsion Б ESP Electrical Power ESP Electrical SLS Electrical Power Power CBS Staging Terminal Propulsion Sequencer CBS ∢

Figure 4.8-10

4-180

$$E_3 = V_3 R_3 + (1-R_3) (K_1 V_3) R_4 + (K_2 V_3) R_3 R_4$$

where

 V_3 = Value of landed experiments

 K_1V_3 = Value given to IDP if Surface Laboratory fails (K_1 = .33)

 K_2V_3 = Value given to IDP if Surface Laboratory is successful (K_2 = .17)

 R_{o} = Reliability of Surface Laboratory equipment

 R_{L} = Reliability of IDP equipment

(1-R₃) = Probability of Surface Laboratory equipment failure

The system effectiveness, landed experiments effectiveness, and improvement in system effectiveness were determined for the two configurations. The results are given in Figure 4.8-11. The addition of an IDP improves both the system effectiveness and effectiveness of landed experiments by .0465. The 100 pounds of improvements in the Flight Capsule increases the System Effectiveness by .0846 and the effectiveness of landed experiments by .0232. Although the IDP gives the best improvement in effectiveness of the landed experiments, it is not effective on a total system basis. The system effectiveness improvement for 100 pounds of improvements is approximately twice that for the IDP. If the IDP equipment reliability and estimated value for the 100 pound IDP could be retained while the weight was reduced the system effectiveness break-even point for an IDP is twenty-five pounds.

It is recognized that the value assigned to K_1 is argumentative. Therefore, an examination of the system effectiveness (E) sensitivity to K_1 was made for the configuration with an IDP. The system effectiveness break even point is at K_1 = 0.94. However, the value of obtaining IDP data (minimum surface data) is clearly not 94% of the value of the Surface Laboratory data.

4.8.5 <u>Conclusions</u> - The results of this analysis are presented with an evaluation of the IDP in Figure 4.8-12.

Installation of an IDP into the Flight Capsule System presents many problems. The size of the IDP severely restricts location within the Flight Capsule. The weight and reaction torque resulting from deployment requires location near (preferably on) the centerline of the Flight Capsule to minimize ballast weight. Installation of the IDP in either location considered, results in inefficient installation of other equipment to provide space for the IDP. A weight penalty, over the IDP weight, is required to install the IDP in the Flight Capsule because of relocating some of the Capsule Bus equipment.

EFFECTIVENESS

ΔΕ	E	E ₃
_	0.7814	0.1715
0.0465	0.8279	0.2180
0.0846	0.8660	0.1947
	- 0.0465	- 0.7814 0.0465 0.8279

$$E = E_1 + E_2 + E_3$$

 E_1 = Effectiveness of achievement of Landing

 E_2 = Effectiveness of achievement of Entry Experiment

 E_3 = Effectiveness of achievement of Landed Experiment.

INDEPENDENT DATA PACKAGE EVALUATION

Favorable Conclusions

- The IDP increases probability of obtaining the surface environmental measurements, gives correlation to the Surface Laboratory (if both operate) measurements, and makes measurements in an uncontaminated area.
- The IDP by-passes four single point failure possibilities, the Terminal Propulsion Subsystem being the only one not improved by adding 100 pounds of improvements to the Flight Capsule.
- The optimum IDP separation point is concept A (approximately 5000 feet). This occurs
 prior to activation of the Terminal Propulsion Subsystem.

Unfavorable Conclusions

- Deployment of the IDP is complicated.
- Installation of the IDP within the Flight Capsule is difficult and would result in an additional weight penalty over the IDP weight.
- Unsuccessful deployment of the IDP, a single point failure, could interfere with the Surface Laboratory experiments and possibly result in total mission failure.
- If the IDP fails to deploy, the S-Band (high rate) antenna cannot be deployed and the T/M data transmission is severely limited.
- The IDP contributes only temperature, pressure, composition, water vapor, wind speed, and some diagnostic measurements. No subsurface or life measurements are made.
- There is development risk associated with the sterilizable high g silver-zinc batteries (20-Wh/lb), impactable 20-watt transmitter, and impactable instruments.
- The uncertainties of successful landing and instrument mast deployment are factors which contribute to a reduction in the total value of the IDP.
- The improvement (100-pounds) to the Flight Capsule gives a greater improvement in probability of success of obtaining the surface environmental measurements than the IDP.
- The 100-pounds Flight Capsule improvement also increases the probability of total mission success, while the IDP makes no contribution to the total mission success.
- The 100-pound improvement to the Flight Capsule increases the landed experiments effectiveness and give approximately twice the improvement in the System Effectiveness as the IDP.

An IDP capability (Configuration 3) or adding the weight as improvements within the Flight Capsule (Configuration 4) are justified based on the reliability analysis. However, the ability of the IDP to by-pass some of the critical single point failure possibilities must also be considered.

The IDP should be released from the Flight Capsule <u>prior</u> to terminal propulsion ignition for maximum mission reliability and maximum independence from critical Flight Capsule single point failure possibilities. IDP release from the Capsule Bus after terminal propulsion ignition results in a very questionable reliability improvement principally because of the criticality of a terminal propulsion engine or landing radar failure.

The IDP is not the most effective method of utilizing excess weight. Adding redundancies into the Flight Capsule improves the probability of total mission success and the overall system effectiveness.

As a result of the installation problems, weight penalties, IDP deployment and interference uncertainties, IDP operational uncertainties after hard landing, and reliability and effectiveness evaluations, it is recommended that the IDP not be incorporated into the baseline design.

However, it is desirable to by-pass the most critical failure possibilities and thereby improve the probability of obtaining some data. The two areas of greatest concern are the surface conditions and the terminal propulsion subsystem. It may be feasible to obtain a better improvement in the probability of collecting some surface data by hardening the Entry Science Package or a portion of the Surface Laboratory. However, the weight penalty should be carefully evaluated to insure effective use of the excess weight.

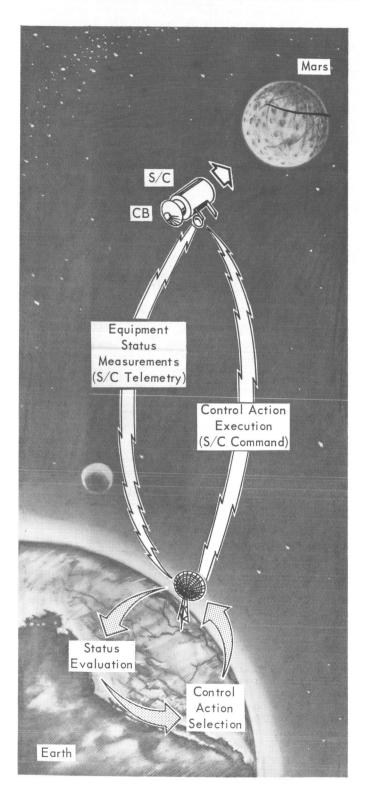
- 4.9 IN-FLIGHT MONITORING AND CHECKOUT A Capsule Bus System in-flight status monitor/checkout/control plan has been developed in parallel with the Capsule Bus design and has been integrated into the overall Mission Support Plan. The automatic monitor/checkout activity includes:
 - a. Continuous passive monitoring from Earth launch through Mars orbit (interplanetary cruise).
 - b. Subsystems activation and performance checkout in Mars orbit prior to Flight Capsule/Spacecraft separation.
 - c. Continuous monitoring of engineering operational parameters from separation through landing (orbital descent).

All of this data is automatically generated and ultimately telemetered by the Space-craft to the Earth stations, where the CB mission operations personnel analyze and judge the integrity and/or performance of the equipment. These same personnel have recourse to corrective/preventive equipment control actions or mission sequence modifications via Earth to Spacecraft command, until Flight Capsule separation from the Spacecraft.

Figures 4.9-1, -2, and -3 present functional descriptions of the status monitor/checkout/control activities for all CB mission phases. Note that data is continuously being gathered on the subsystems; cruise parameter monitoring continues in orbit both before and after the pre-separation checkout period. The methods of all data reception, distribution, and analyses by the mission operations personnel at the Deep Space Instrumentation Facility and Space Flight Operations Facility are discussed in detail in Section D 4.5.

- 4.9.1 <u>Test Purpose and Selection Criteria</u> The purpose of in-flight monitoring and checkout is to maximize the probability of mission success. Whether or not mission objectives or operations will be changed prior to separation will depend on the condition of the Capsule Bus equipment, as determined from the monitor and checkout data. The test selection criteria establish the tests necessary to perform this function with proven engineering techniques.
- 4.9.1.1 <u>Interplanetary Cruise Monitoring</u> Continuous passive monitoring of equipment temperatures and pressures and the thermal control, propulsion, and electrical power subsystem's status are required to maintain confidence of CB survival during its inactive transit to Mars. The confidence level of the received data is established by having the telemetry unit also monitor the operating conditions of itself. These data allow early detection of impending problem conditions or failures and

CAPSULE BUS INTERPLANETARY CRUISE STATUS MONITOR/CONTROL PLAN



Equipment Status Measurements

• Continuous Passive Monitoring

Status Evaluation (Mission Operations Personnel)

- Flight Systems Integrity Verification
- Impending Failure or Failures
- Ground Equipment Accuracy Verification

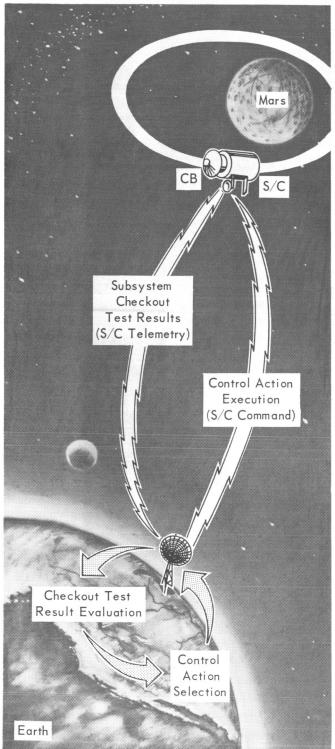
Control Action Selection (Mission Operations Personnel)

- Choose Subsystem Corrective/ Preventive Action
- Select Mission Contingency Plan
- Check Ground Equipment

Control Action Execution

 Send Command to Spacecraftto-Capsule Bus

CAPSULE BUS PRE-SEPARATION CHECKOUT TEST/CONTROL PLAN



Subsystem Checkout Test Results

- Subsystems Mission Simulated Inputs and Outputs Monitored
 - •• Test Sequence Controlled Automatically by Capsule Bus Test Programmer

<u>Checkout Test Result Evaluation (Mission Operations Personnel)</u>

- System Operation/Calibration Verified
- Fault Isolation
- Ground Equipment Accuracy Verification

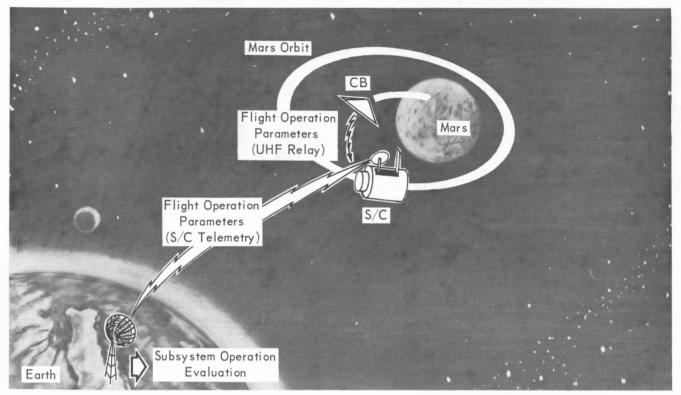
Control Action Selection (Mission Operations Personnel)

- Change Subsystem Configuration/Mode
- Modify Mission Sequence and/or Profile
- Repeat Particular Checkout Test

Control Action Execution

- Send Command to Spacecraft-to-Subsystem
- Send Command to Spacecraft-to-Test Programmer

CAPSULE BUS ORBITAL DESCENT AND ENTRY MONITOR PLAN



Subsystem Flight Operation Results (UHF Radio Relay and S/C Telemetry)

• Continuous Monitor of Capsule Bus Flight Parameters

Subsystem Operation Evaluation (Mission Operations Personnel)

- Record Data for Post-Flight Analyses
- Cursory Flight Dynamics Analysis to Update Entry Profile of Second Flight Capsule.

guide the ground personnel in selecting the best command control action.

4.9.1.2 <u>Pre-Separation Checkout Tests</u> - Prior to the Flight Capsule/Spacecraft separation, the CB equipment will be activated and tested under simulated mission inputs (where practical). The test sequence and equipment operation is under the control of an on-board, pre-programmed, automatic test programmer. Equipment operational parameters and test responses are evaluated by the mission operations personnel and are used to:

- a. Determine the operational performance of CB subsystems prior to mission commitment.
- b. Isolate faults to the subsystem module level.
- c. Select redundant components or functional backup modes of operation.
- d. Modify mission profile or event sequence.
- e. Provide correlation data to facilitate post-flight analyses and compare with pre-launch calibration data.

The test details are constrained by the requirements for minimizing reliability degradation, test equipment complexity, and consumption of mission power as a result of testing.

An extensive equipment built-in self-test capability is required for remote pre-launch checkout after Flight Capsule sterilization. Many of the same equipment capabilities will be employed again to attain considerable in-flight test depth.

4.9.1.3 Mission Operation Monitoring - The equipment operational parameters monitored during mission operation are largely the same as those monitored during the dynamic checkout tests, which utilized simulated mission inputs. Typical additions include the pyrotechnic actuation events. The sampling rates of some of the parameters are increased during post-separation flight to detect transient conditions over wider ranges of inputs. These data will be used to:

- a. Develop a discrete record of vehicle flight dynamics.
- b. Determine any desirable equipment modifications (i.e., response time, dynamic range, etc.) to increase control sensitivity and capability on future missions.
- c. Isolate fault causes between the cruise and atmospheric entry environments. The ground personnel will use these parameter data to perform a cursory flight response evaluation and determine desirable mission updates for the second Flight Capsule; the Entry Science Package atmospheric data measurements will also aid in selecting the updates.
- 4.9.2 <u>Subsystem Design Implications</u> The decision to include the continuous interplanetary cruise monitoring capability has resulted in the design of a special

purpose, hard-wired telemetry commutator and subsystem interfaces with this unit. The cruise commutator is a low power, high reliability unit using proven design techniques. The equipment telemetry transducers are simple in design and low in number for this function.

The decision to include the on-board dynamic subsystems checkout test capability has resulted in a parallel design requirement for subsystem compatibility with the required synthetic test input stimuli, the test programmer, and the telemetry subsystem. Due to this early integration effort on the test planning, the equipment designers have played an important role in deciding on each of the test parameters, the special test equipment design, and interface definitions. on-board, automated test capability is also required for remote pre-launch equipment checkout after Flight Capsule sterilization. The equipment test stimuli generators have been chosen according to each selected pre-launch and in-flight test on each element and are self-contained within the primary equipment. internal packaging concept has been chosen to minimize equipment/test stimuli design compatibility and integration problems. The CB equipment are required to interface with the Capsule Bus test programmer, which automatically commands the subsystems into their test modes and cycles the test stimuli according to a preprogrammed event/time schedule (test sequence). The data requirements to evaluate the equipment test responses and validate the proper test stimuli and test programmer outputs have been included in the analyses to determine the telemetry subsystem modes and capacity.

The CB telemetry subsystem has been designed to operate in three different modes during the post-separation period; these are the de-orbit, entry, and terminal descent modes. This monitoring mode sequence is used to eliminate transmission of equipment operational parameters during period when they are inactive (e.g., the landing radar and terminal propulsion subsystems are not activated until approximately 2 minutes prior to touchdown).

4.9.3 <u>In-Flight Monitor and Checkout Test Descriptions/Discussion</u> - Figure 5.5-2 of Section B5.5 (the CB telemetry instrumentation list) presents the data which are transmitted to Earth during the CB mission phases. Also shown are the accuracy of these measurements and their sampling rates. These 5 different data acquisition modes provide continuous CB subsystem data; the cruise data continues in Mars orbit both before and after the pre-separation checkout tests. For the reason stated above, note the post-separation monitoring sequence of: (1) de-orbit, (2) entry, and (3) terminal descent. For this flight phase, note also the inclusion of the

monitoring of all pyrotechnic actuation events.

Figure 4.9-4 presents the functional description and test objective for the dynamic checkout tests performed on each subsystem during the pre-separation phase. The actual subsystem parameter data gathered during these tests is listed in Figure 5.5-2 of Section B5.5 under the heading of pre-separation checkout. The corrective actions taken as a result of the test data for a subsystem, as listed in Figure 4.9-4, are cited as examples. All of the test results must be evaluated before the corrective action plan, if any, is performed.

All CB checkout tests are conducted in a pre-programmed, automatic step sequence for 2 hours and 2 minutes, starting 24 hours prior to separation from the Space-craft. The Entry Science Package and Surface Laboratory tests are conducted sequentially after the completion of the CB tests. This integrated test phasing and timeline are designed to allow adequate time for selecting any desirable mission updates. Moreover, the test results of one system can affect the mission decisions for other systems; for example, if ESP entry TV fails, Flight Capsule separation can be delayed and the other capsule landed first. Also, test re-runs might be selected for any of the systems. The selected time margin from test initiation to planned separation is ample to decide on and implement the optimum available overall (Capsule Bus, Entry Science Package, Surface Laboratory) mission plan.

All the checkout tests (except the terminal propulsion throttable rocket response check) are performed on Spacecraft power over a 2 hour period; the average test power level is 155 watts with a peak of 198 watts. The 2 hour period is determined by a 1 hour IMU warm-up period followed by a 1 hour drift rate test. The other tests are run concurrently in such a manner as not to exceed the 200 watt Spacecraft power limitation. The terminal propulsion throttable rocket response test takes 2 minutes on internal CB battery power at a 351 watt level. This test consumes less than one percent of the CB battery capacity; this energy is restored after testing by putting the battery back into the charge mode.

The in-flight checkout test duty cycle and turn on/off reliability considerations for each subsystem have been included in the overall CB mission reliability calculations. It has been determined that the change in probability of equipment failure of the CB is less than three percent as a result of these test operations. This change is insignificant relative to the mission success enchancement available as a result of the test information and command capability to select redundant components and functional back-up modes of operation.

CAPSULE BUS SUBSYSTEMS PRE-SEPARATION CHECKOUT TESTS

TEST NAME	TEST DESCRIPTION
Guidance and Control Subsystem 1. IMU Drift Rate Test 2. Accelerometer and De-Orbit ΔV Cutoff Check	 G & C computer integrates attitude signals from gyros over a one hour period when spacecraft limit cycles in a sun/canopus attitude hold mode. Calibrated current source applied to auxiliary torquer of pulse rebalance accelenter. Monitor computer integrated acceleration (△V) level at which computer generate de-orbit motor cutoff signal.
3. De-Orbit Maneuver Sequence Check 4. Computer Algorithms Check	 Computer internally commands precision Eulerian angular rate over precise time terval to provide maneuver angular displacement command. Internal check of program instructions, arithmetic logic, and memory words such maneuver angles, de-orbit ∆V, terminal descent R/V points, etc.
Radar Subsystem 1. Altimeter Calibration and Logic Check	 The modulator is time delayed from a nominal operational sync pulse and the receiver is allowed to lock-up on a calibrated RF leakage from the transmitter The receiver verifies 200,000 feet altitude measurement and logic mark The above test is repeated on the secondary antenna Switched to short range mode and above tests are repeated
2. Landing Radar Calibration	 Built-in self-test circuitry/oscillators simulate ground returns to range and dopy velocity trackers (both IF to pre-amplifiers and RF to mixers) to calibrate receit tracker subassembly. Doppler and range transmitters are turned on and antenna radiates into RF absocap to simulate free space conditions — output powers and range beam modulator waveform measured.
UHF Relay Radio & Antenna Subsystem	
Mars-to-Earth Verification of Capsule Bus Low Rate Link	 Simulated "test words" cause transmitter modulation and the low rate antenna radiates RF into a special on-board parasitic antenna. The parasitic antenna received signal is coupled into spacecraft receiver, behi the receiving antenna, by a coaxial directional coupler. The spacecraft received "test words" is transmitted to earth by the spacecraft
Sequencer and Timer Subsystem 1. Operational Check	 Turn on and run stored program in fast time — all output drivers are exercised Test programmer functions as OSE during test to hold all squib circuit outputs the "safe" test condition.
2. Memory Check and Update	Readout memory and update as required Readout memory after update (if update action performed)

Figure 4.9-4

TEST OBJECTIVE/CORRECTIVE ACTION/SPECIAL REMARKS

- Computer updates its gyro drift rate compensation words in memory.
- Verifies accelerometer performance for de-orbit thrust cutoff and terminal descent deceleration thrust command function (same accelerometer for both functions)
- ullet Bias the de-orbit ΔV word in the computer memory
- Provide de-orbit motor cutoff signal by time on/off from Capsule Bus Sequencer and Timer
- Verifies De-Orbit maneuver capability and accuracy
- With no maneuver command capability abort mission and do not separate
- Verifies operational status and guidance parameters.
- Telemetry parameters determine altitude measurement accuracy, verify altitude mark and range mode change, and allow fault isolation.
- With altimeter malfunction the G&C subsystem computer integrated ΔV can be used as backup for aerodynamic decelerator deployment
- With failure, science loses altitude histogram used in determing atmospheric properties
- Radar range/velocity measurements verify landing radar accuracy.
- Engineering telemetry parameters allow fault isolation of both the transmitting and receiving subassemblies.
- Earth reception of "test word" verifies CB/spacecraft/earth relay performance, except for integrity of spacecraft mounted receiving antenna.
- Telemetry parameters allow fault isolation of transmitter and receiver compatibility problems
- Telemetry cross-strap techniques include redundant path for all CB low rate data from entry to landing on Entry Science package high data rate link.
- Test verifies proper time generation and all output circuit closures.
- This test verifies memory retention during cruise.
- The update words depend in many instances on results of other checkout tests
- Memory readout after update verifies proper update.

CAPSULE BUS SUBSYSTEMS PRE-SEPARATION CHECKOUT TESTS (Continued)

TEST NAME	TEST DESCRIPTION			
Propulsion Subsystem 1. Terminal Throttable Rocket Response Check	 Internal G&C computer routine commands each solenoid valve open and applie multi-level commands to throttle servos — jet fire inhibited. Solenoid valve position and servo feedback valve positions monitored. Jets are tested one at a time on internal power — test power requirements excesspacecraft power limitation — automatically activated batteries for these high discharge rate elements are not active. 			
Thermal Control Subsystem 1. Temperature and Heaters Check	The same equipment temperatures and heater currents as monitored during integration planetary cruise are also monitored during in-flight checkout.			
Electrical Power Subsystem 1. DC/DC Converter Performance 2. Batteries Condition Check	 The same DC/DC converter temperature and output voltage/current as monitoring cruise are also monitored during in-flight checkout test. The same temperature and open/closed circuit output voltage/current and char current as monitored during cruise are also monitored during checkout tests. The batteries are under load only for the Terminal Rocket Response Check Terminal Rock			
Telemetry Subsystem 1. Linearity Test	 Calibration voltages are applied to all low level amplifiers and A/D converter monitored Multi-level reference signals provide linearity (accuracy) test of these elementheir entire input range. 			
Data Storage Subsystem 1. Data Delay Function Check 1. De-Orbit Cruise and Thrust Attitude Control Function Check (IMU, G&C Computer, RCS Jets)	 All inflight checkout test data is also delayed transmitted by 50 and 150 seco Calibrated current source is applied to auxiliary torquer of each gyro. Computer outputs to RCS jets are checked for both the attitude hold and deriv rate modes. RCS jet on/off response monitored-(jet fire inhibited). 			

Figure 4.9-4 (Continued)

	TEST OBJECTIVE/CORRECTIVE ACTION/SPECIAL REMARKS
ed .	 Verifies capability of throttable rocket start-up. Multi-level reference inputs from computer provide linearity (accuracy) check on throttle servo positioning transducers over entire thrust range.
·r-	Normal heater power from spacecraft is used now for checkout tests.
ed dur- ging	 Verifies DC/DC converter performance Switch to redundant DC/DC converter Verifies battery charge state, cell status, and detects self discharge. If other systems test data show anomalies, this data allows correlation for isolation of fault cause (i.e., under specification voltage, wrong power distribution)
outputs ts over	 Test data indicates how to bias interpretation of received subsystem data An abbreviated version of this test is also performed during flight to provide confidence level of flight information.
hds.	Verifies data delayed transmission function for blackout regime data Verifies IMU/computer/RCS jets end-to-end response for both attitude hold and derived rate modes.

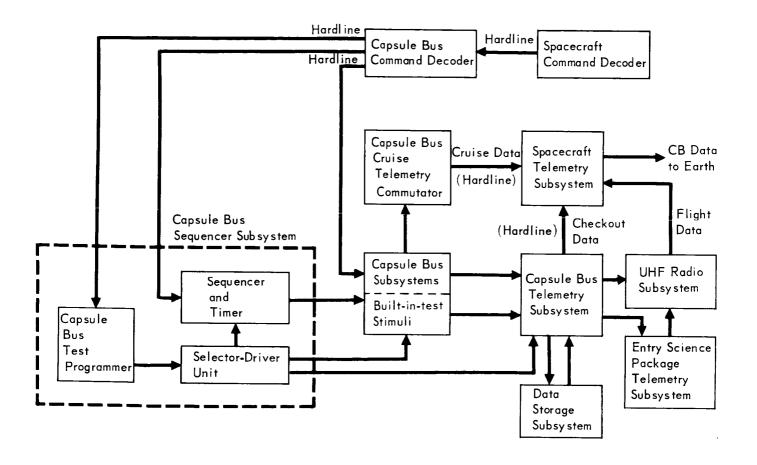
- 4.9.4 <u>Monitor/Checkout Test Data/Command Interfaces</u> The data generation/gathering techniques and rates differ, as does the command interface to perform status control for each of the CB mission phases. Figure 4.9-5, presents a functional data/command interface description of the Capsule Bus and Spacecraft equipment for all mission phases.
- 4.9.4.1 <u>Data Interface Description</u> From launch to the pre-separation checkout tests, the cruise commutator is the only CB data source. This data is hardlined to the Spacecraft, which transmits the data to Earth.

The CB test programmer outputs result in accomplishing the tests of Figure 4.9-4 according to the pre-programmed 2 hour and 2 minute schedule as discussed in the last section. A detailed description of this test programmer is given in Section C 8.2. The test programmer first commands the CB telemetry subsystem into the checkout mode to select all the checkout test parameter channels. The test programmer then sequences the subsystems on and appropriately cycles the internal equipment test stimuli (i.e., turns on "test word" generator, cycles calibrated current source inputs to gyros and accelerometers, etc.). The test programmer outputs are time tagged commands which are decoded by the Selector-Driver Unit and result in either Sequencer and Timer or Selector-Driver Unit output circuit closures. The test programmer output time delay between successive steps in any one test are based on estimated times for equipment stabilization; for example, the gyro spin motor power is applied one hour after gyro heaters turn on. All the checkout data is sent via hardline from the CB telemetry subsystem to the Spacecraft telemetry subsystem, which controls transmission to the Earth stations.

The CB telemetry subsystem selects different information channels during the mission flight (de-orbit, entry, terminal descent) to prevent interrogating inactive subsystems. All data after entry (800,000 feet altitude) is also delayed 50 seconds and 150 seconds by the data storage subsystem; this prevents loss of CB data during entry communications blackout. All CB data after entry is also interleaved and repeated by the ESP telemetry subsystem; this provides a redundant CB data path. The total CB telemetered outputs are relayed to the Spacecraft by the UHF radio subsystem, and the Spacecraft transmits the data to Earth stations.

4.9.4.2 <u>Command Interface Description</u> - The CB mission is designed to be automatic from launch to landing. However, a command change capability is designed to safeguard its status and/or modify its mission. Figure 6-5 of Section C 6 presents the CB command control capability to exercise the subsystems or update subsystem memories for mission sequence changes. The commands from Earth to the Spacecraft

CAPSULE BUS IN-FLIGHT TEST DATA AND COMMAND INTERFACE DIAGRAM



are hardlined to the CB subsystems as shown in Figure 4.9-5.

Special note is given that the test programmer is capable of being updated inflight. This allows test routine time changes between successive test steps and the repeat of any single test. This capability is required to allow equipment stabilization during checkout tests, if first test data shows pre-programmed estimated times inadequate. The test programmer can also be employed to increase equipment temperature during cruise, if so desired, by turning equipment on with subsequent power dissipation and warming. If needed, an early checkout test routine can be initiated via command to the test programmer (hardline from Spacecraft).

There is no CB command capability after Flight Capsule separation from the Spacecraft.

4.10 SYSTEM EFFECTIVENESS ANALYSIS - Proper allocation of weight available for reliability improvement has been an important factor in our Flight Capsule studies. Our preferred design includes redundant components weighing 73 pounds. Six items (weighing 43 pounds) were included because our engineering judgement indicated they were necessary to satisfy design constraints. An additional 21 items were selected from a list of 45 recommended as a result of our system effectiveness analysis. This technique identifies redundancies which yield the maximum effectiveness gain per unit weight. Weight limitations, development risk, and potential subsystem integration problems are among the reasons for not including the other 24 at this time. Additional redundancy could be achieved by utilizing a portion of the Flight Capsule's 220 pound weight margin. Use of redundancy in our preferred design has increased, to 71%, the probability that all Flight Capsule equipment (excluding experiments) will function properly. (See Figure 4.10-1) The estimated reliability of the capsule (excluding experiments) for achieving landing is increased to 87% and for performing entry experiments is increased to 86%. 4.10.1 Technique - The system effectiveness analysis is used to evaluate redundancies in terms of reliability improvement, change in weight and mission objectives. We used the effectiveness equation $E = V_1 R_1 + V_2 R_2 + V_3 R_3$ in our calculations. Each term of this equation represents a single mission objective. The first term, ${f v}_1^{}$ ${f R}_1^{}$, represents "Achievement of a Flight Capsule Landing". ${f v}_1^{}$ is the value assigned to the event, and $R_1^{}$ the estimated reliability of the Capsule Bus for performing that event. The second term, V_2 R_2 , represents "Performance of Entry Experiments". V_2 is the value assigned to the performance of entry Science experiments and $R_{2}^{}$ is the estimated reliability of the Flight Capsule (excluding experiments) to complete that phase of the mission and transmit the data. The third terms, ${
m V_3}$ ${
m R_3}$, represents "Performance of Landed Experiments". ${
m V_3}$ is the value assigned to the performance of landed experiments; R_3 is the estimated reliability of the Flight Capsule (less experiments) to perform landed science experiments. Derivation of this method is described in Reference 4.10-1. 4.10.1.1 <u>Value</u> - Mission objectives, in order of decreasing importance, are listed in Reference 4.10–2. This ordering was used in the selection of values such that $v_1>v_2>v_3$. Various value distributions (with $v_1+v_2+v_3=1$) were used to determine the influence of value assignments in the effectiveness analysis. Significant effect of value assignment occurred only when V_1 , the value assigned to landing, exceeded 0.50. For our current analysis we have used:

FLIGHT CAPSULE RELIABILITY

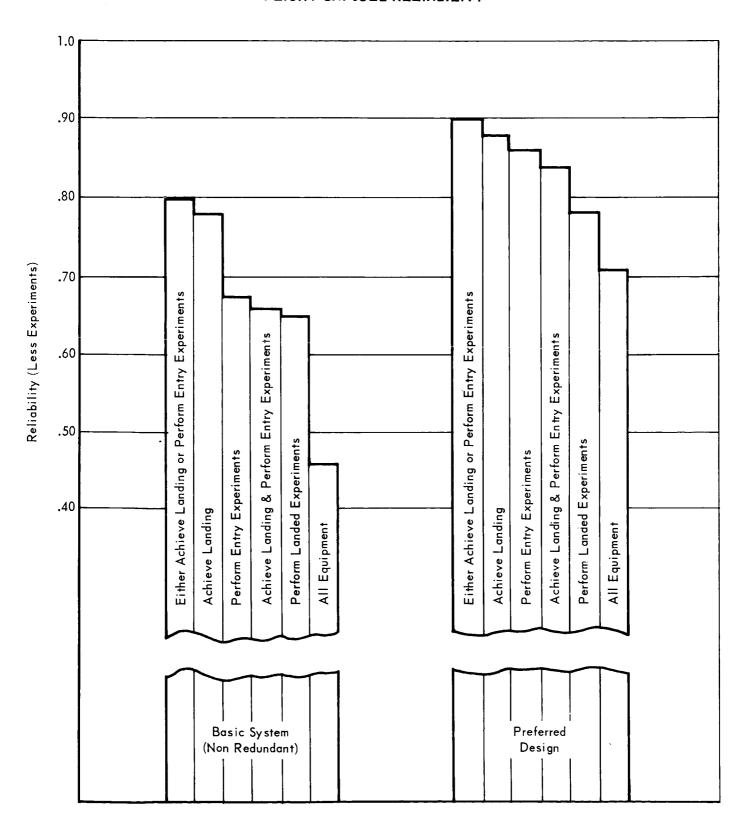


Figure 4.10-1

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 V_1 (Successful landing) = 0.40

 V_2 (entry experiment) = 0.35

 V_3 (landed experiments) = 0.25

This approach to system effectiveness emphasizes the reliability of the components which are essential to completion of two or more of the objectives. Further, it de-emphasizes component redundancy where a component is used for a single objective or multiple paths are available. The reliability of the Flight Capsule components required to accomplish each objective enters the computation in conjunction with the value of that objective. The subsystem components (Figure 4.10-2) - CB sequencing, staging, reaction control and de-orbit propulsion, guidance and control, thermal control and electrical power - are required to operate to perform any of three events. A value of 1.00 is associated with these components since their reliability enters all three terms of the equation. Terminal propulsion and landing components operations are required to perform the first and third events: value ${f v}_1$ + ${f v}_3$ was associated with these components. Successful operation of any one of the three parallel paths (see the lower part of Figure 4.10-2) of telecommunications will provide verification of a successful landing. Therefore, the resultant reliability of this parallel combination was assigned the value, \mathbf{V}_1 . 4.10.1.2 Weight Factors - The weight increment directly associated with each redundancy is that of the component itself. However, incorporation of each redundancy imposes an additional weight increment to support that component. For example, the addition of component to the Surface Laboratory System requires a corresponding increase in the weight of those Flight Capsule subsystems required to land that component on the surface. The terminal propulsion subsystem would require more fuel, the parachutes would be a little larger, and so on. This effect was included in the analysis by using weight factors. The weight factor is the partial derivative of the Flight Capsule weight with respect to component weight. In this analysis we have used a difference ratio to approximate the derivative as indicated in Figure 4.10-3. For example, a recommended redundant relay, actual weight 0.3 pounds, added to the Surface Laboratory electrical power subsystem would cause a resultant weight addition of 0.54 pounds to the Flight Capsule weight:

$$\Delta W_{FC} = W_A \times F_{SL}$$

$$\Delta W_{FC} = .3 \times 1.81$$

$$\Delta W_{FC} = .54$$

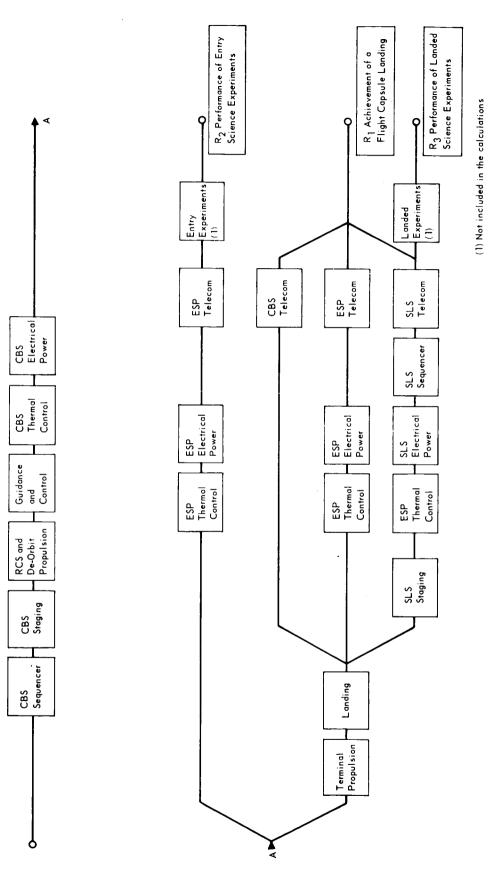


Figure 4.10-2

WEIGHT FACTORS

		MISSION PHASES					
	FLIGHT CAPSULE	PRE DE-ORBIT	POST DE-ORBIT	ENTRY	TERMINAL DESCENT	TOUCH DOWN	SURFACE LAB
5500 lb Capsule (1)(2) 4500 lb Capsule (1)(3) AW (4)	5500.0 4500.0 10 00.0	4678.2 3684.5 993.7	4169.9 3283.6 886.3	4084.7 3214.6 870.1	3164.6 2396.1 768.5	3089.6 2321.1 768.5	1159.6 607.7 551.9
Weight Factor (5)	1.00	1.01	1.13	1.15	1.30	1.30	1.81

- (1) VOYAGER Weight Program Computer Run 22 June 1967 (Not Published)
- (2) 5500 lb Weight Summary Values
- (3) 4500 lb Weight Summary Values
- (4) ΔW Difference in Mission phase weight (2) –(3)

(5) Weight Factors =
$$\frac{1000}{\Delta W}$$

where:

 ΔW_{FC} = The added Flight Capsule weight to accommodate the redundancy

 W_{A} = The actual weight of the added component

 $F_{\rm SL}$ = The weight factor associated with components added to the Surface Laboratory

4.10.1.3 <u>Selection</u> - The selection of recommended items of redundancy is made by evaluating the ratio of the weight addition to the change in the effectiveness. The magnitude of this ratio, $\Delta W/_{\Delta E}$, established the order of selection. The component with the losest $\Delta W/_{\Delta E}$ is selected first, followed by components of increasing $\Delta W/_{\Delta E}$. This selection is similar to a reliability analysis. (See Section E2) We applied the effectiveness analysis in recommending items to be made redundant for reliability improvement.

A portion of the effectiveness analysis results is shown in Figure 4.10-4 with a similar portion of the reliability analysis for comparison. A guidance and a propulsion redundancy, G10C and P1C, were selected for a specific example of the differing results which would be obtained from the two approaches:

G10C - Active redundant receivers and trackers in radar altimeter

P1C - Redundant cartridge in each of three normally closed RCS pyro modules The reliability analysis, as shown in Figure 4.10-4, indicates the desirability of incorporating the guidance redundancy, G10C if up to 100 pounds of redundancy were allowable. Effectiveness analysis gives higher priority to that redundancy and indicates inclusion of this redundancy if only 70 pounds of redundancy were allowable. A similar example is shown with P1C, the propulsion component redundancy. The reliability analyses method shows that 169 pounds must be available to warrant adding the redundant cartridges; the comparable threshold is 89 pounds using the effectiveness analysis.

4.10.2 <u>Final Selection</u> - Satisfying the design constraints was considered most important in the final selection. Six elements were made redundant because our engineering judgment indicated they were necessary to satisfy the Flight Capsule design constraints. These included two items in the Capsule Bus. A third squib battery (2 of 3 required) will improve the probability of providing adequate power to high current solenoids and valves in the propulsion system. Dual shielded mild detonating cord is provided for devices used in forward canister release. The systems effectiveness analysis was used as a guide in selecting redundancies for the preferred design. The first 45 redundancies listed in Figure 4.10-4 were considered for the preferred Flight Capsule design, and 21 were selected. Sixteen of these

RECOMMANDED COMPONENT REDUNDANCY

RELIABILITY ANALYSIS								
REDUNDANCY CODE (1)	ΔW/ Δ Ln R	RELIABILITY (2)	REDUNDANCY WEIGHT (3)	REDUNDANCIES	REDUNDAI CODE (1)			
BASIC SYSTEM	_	.4579			BASIC SYS			
TIIC	9	.4680	.19	STANDBY REDUNDANT CBS CRUISE ENCODER	T 15E			
T 15E	12	.4757	.39	STANDBY REDUNDANT ESP CRUISE ENCODER	TIE			
T10C	12	.4991	.97	SERIES ACTIVE REDUNDANT CBS CRUISE COMMUTATOR DATA SWITCHES & DRIVERS	T 14E			
TIE	15	.5101	1.30	INTERLEAVE LOW RATE ESP DATA ON CBS RADIO LINK	T23S			
T 23S	16	.5185	1.57	STANDBY REDUNDANT SLS CRUISE ENCODER	(7) ¦ E1A			
T 14E	18	.5357	2.15	SERIES ACTIVE REDUNDANT ESP CRUISE COMMUTATOR DATA SWITCHES & DRIVERS	T22S			
T2C	20	.5447	2.48	INTERLEAVE LOW RATE CBS DATA ON ESP RADIO LINK	Elic			
T 22S T 37C	22	.5655	3.29	SERIES ACTIVE REDUNDANT SLS CRUISE COMMUTATOR, DATA SWITCHES & DRIVERS	TIIC			
T12C	28	.5749	3.68	SERIES ACTIVE REDUNDANT CBS CRUISE MONITOR CONTROL DATA SWITCHES & DRIVERS	T 10C			
E1A	50	.5875 .6626	4.28 10.26	REDUNDANT ADAPTOR CRUISE COMMUTATOR AND CRUISE ENCODER	E23C			
ELIC	165	.6788	14.26	CBS & ESP BATTERY REDUNDANCY PROVIDED BY THE SLS BATTERY ACTIVE REDUNDANT DC-DC CONVERTER REGULATORS	(7) — \$1C			
E 17S	245	.6804	14.20	ACTIVE REDUNDANT BOTTERY CHARGER RELAY	G1C E19C			
E 18S	245	.6819	15.34	ACTIVE REDUNDANT BATTERY CHARGER RELAY	E19C			
E 15S	245	.6834	15.89	ACTIVE REDUNDANT BATTERY CHARGER RELAY	E13C			
E 16S	245	.6849	16.43	ACTIVE REDUNDANT BATTERY CHARGER RELAY	1 T37C			
E16E	287	.7006	22.94	STANDBY REDUNDANT BATTERY FLOAT CHARGERS	(7) { G8C			
E23C	287	.7167	29.44	PROVIDE STANDBY REDUNDANT BATTERY FLOAT CHARGERS	T12C			
\$1C	347	.7176	29.86	REDUNDANT REEFING CUTTERS FOR EACH PARACHUTE REEFING LINE (1 OF 3 REQUIRED)	G3C			
T8C	356	.7233	32.72	STANDBY REDUNDANT CBS COMMUTATOR AND ENCODER	E16E			
GIC	365	.7334	37.76	MULTI AXIS GYRO SENSING	(7)— G10C			
E 18C	440	.7350	38.73	ACTIVE REDUNDANT COMMAND DECODER RELAYS	E 17S			
E 19C	440	.7366	39.71	ACTIVE REDUNDANT CRUISE COMMUTATOR RELAYS	E 18S			
E 13C	441 441	.7383	40.68	ACTIVE REDUNDANT BATTERY CHARGER RELAYS	E 155			
E12E T20E	478	.7399 .7450	41.66 44.91	ACTIVE REDUNDANT BATTERY CHARGER RELAYS	E 16S			
T 17E	525	.7450	44.91	ACTIVE REDUNDANT ESP TV BUFFER ACTIVE REDUNDANT ESP TV DATA PROCESS ELECTRONICS	H2C			
T7C	567	.7477	47.78	STANDBY REDUNDANT CBS PROGRAMMER	(7) T2C S2C			
G8C	581	.7574	54.28	FOUR LANDING RADAR VELOCITY SENSOR CHANNELS (3 REQUIRED)	E12E			
7 28S	611	.7619	57.91	STANDBY REDUNDANT SLS TV DATA PROCESS	G2C			
T 16E	649	.7631	58.95	ACTIVE REDUNDANT ESP SCIENCE DATA REMOTE INTERFACE ELECTRONICS	C29C			
T 18E	660	.7664	61.81	STANDBY REDUNDANT ESP COMMUTATOR AND ENCODER	T20E			
G3C	676	.7919	83.93	ONE OF TWO G AND C COMPUTERS SELECTED DURING IN-FLIGHT CHECKOUT	(7) S3C			
T 13E	738	.7929	84.84	STANDBY REDUNDANT ESP PROGRAMMER	\$5C			
T24S	819	.8015	93.72	STANDBY REDUNDANT SLS COMMUTATOR AND ENCODER	` P1C			
T265	859	.8024	94.62	STANDBY REDUNDANT SLS CONVOLUTION CODER	(7) — S6C			
G 10C	900	.8074	100.22	ACTIVE REDUNDANT RECEIVERS AND TRACKERS IN RADAR ALTIMETER	C46C			
T36S	962 987	.8106	104.02	ACTIVE REDUNDANT SLS COMMAND SUBSYSTEM DECODER	T 17E			
T30S H3E	1055	.8256	122.14	FUNCTIONAL REDUNDANT SLS TAPE RECORDER STORAGE	C42C			
H3E H2C	1101	.8259 .8273	122.49 124.45	ACTIVE REDUNDANT THERMOSTATS ACTIVE REDUNDANT THERMOSTATS	(7) — S4C			
T21S	1184	.8305	124.45	STANDBY REDUNDANT SLS PROGRAMMER	T 16E			
\$2C	1241	.8325	131.96	DUAL CARTRIDGE EXPLOSIVE BOLTS - CAPSULE BUS ADAPTER SEPARATION	T 18E P2C			
G2C	1260	.8349	135.60	ONE OF TWO ACCELEROMETERS AND ELECTRONICS SELECTED DURING CHECKOUT	T 13E			
T5S	1312	.8422	147.01	FUNCTIONAL REDUNDANT SLS LOW RATE RADIO LINK	(7) — G9C			
T3S	1318	.8533	164.23	FUNCTIONAL REDUNDANT SLS MONOPULSE TRACKING	HIC			
C29C	1328	.8534	164.36	ACTIVE REDUNDANT CRYSTAL CONTROLLED OSCILLATORS. (CBS)	C110			
C24S	1349	.8535	164.53	ACTIVE REDUNDANT CRYSTAL CONTROL OSCILLATORS (SLS)	T 28S			
\$ 3C	1391	.8545	166.20	DUAL CARTRIDGE EXPLOSIVE BOLTS - DEORBIT MOTOR RELEASE	G4C			
S5C	1417	.8555	167.91	DUAL CARTRIDGE EXPLOSIVE BOLTS AEROSHELL RELEASE	C37C			
PIC	1436	.8560	168.77	REDUNDANT CARTRIDGE IN EACH OF 3 N.C. PYRO VALVES (RCS)	H3E			
S6C	1436	.8562	169.05	REDUNDANT INITIATORS IN PARACHUTE CATAPULT	T24S			

(1) The first letter in the column identifies the subsystem.

C — Sequence and Timer L — Landing H
G — Guidance and Control P — Propulsion T
E — Electrical Power S — Staging (Separation

H - Thermal Control
T - Telecommunications S-Staging (Separation, deployment & release devices)

The letter following the number identifies the system.

A — All three systems E — Entry Science Package
C — Capsule Bus System S — Science Laboratory System

Figure 4.10-4

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(2) Doe

(3) Incl (4) Ach

EFFECTIVENESS ANALYSIS									
CY	.\₩/ \ E	EFFECTIVENESS E · V1R1+ V2R2+ V2R2	REDUNDANCY WEIGHT	RELIABILITY (2)		(2)	REDUNDANCIES		
		1.1. 2.2 13.3	(3)	(4)	(5)	(6)			
EM	-47 59 71 94 99 122 229 286 399 489 511 618 620 763 814 884 139 1264 1406 1406 1406 1406 1548 1565 1767 1771 1869 1914 2022 2022 2022 2028 2028 2028 2028 202	F - V ₁ R ₁ + V ₂ R ₂ - V ₃ R ₃ .7103 .7145 .7200 .7286 .7315 .7941 .8013 .8209 .8217 .8233 .8422 .8432 .8549 .8568 .8587 .8607 .8612 .8709 .8717 .9007 .9084 .9141 .9146 .9151 .9155 .9160 .9177 .9179 .9201 .9209 .9236 .9236 .9236 .9236 .9238 .9290 .9310 .9323	WEIGHT (3) 19 .52 1.10 1.37 7.34 8.16 12.16 12.35 12.94 19.45 19.86 24.90 25.87 26.85 27.83 28.22 34.72 35.32 57.44 63.95 69.54 70.09 70.63 71.17 71.72 73.67 73.99 76.97 77.95 81.59 81.72 84.97 86.64 88.35 89.21 89.49 92.66 94.61	(4) .7793 .7798 .7805 .7815 .7820 .8311 .8324 .8528 .8547 .8588 .8785 .8917 .8937 .8997 .8997 .8992 .9093 .9113 .9416 .9424 .9483 .9485 .9485 .9486 .9503 .9509 .9532 .9533 .9561 .9564 .9576 .9587 .9593 .9593 .9593	.6761 .6873 .7025 .7256 .7256 .8196 .8196 .8398 .8398 .8398 .8398 .8590 .8601 .8720 .8739 .8759 .8778 .8778 .8778 .8778 .8877 .9172 .9383 .9441 .9441 .9441 .9441 .9441 .9441 .9441 .9441 .9458 .9458 .9458 .9458 .9502 .9529 .9529 .9530 .9595 .9607 .9618 .9624 .9624 .9624 .9626 .9647 .9683	.6480 .6480 .6480 .6480 .6587 .6990 .7258 .7436 .7436 .7436 .7436 .7607 .7616 .7773 .7773 .7773 .7773 .7773 .7861 .8122 .8122 .8122 .8122 .8122 .8122 .8122 .8122 .8122 .8123 .8297 .8246 .8260 .8280 .8280 .8280 .8304 .8305 .8305 .8315 .8325 .8330 .8331 .8349 .8349	STANDBY REDUNDANT ESP CRUISE ENCODER INTERLEAVE LOW RATE ESP DATA ON CBS RADIO LINK SERIES ACTIVE REDUNDANT ESP CRUISE COMMUTATOR DATA SWITCHES & DRIVERS STANDBY REDUNDANT SLS CRUISE ENCODER CBS & ESP BATTERY REDUNDANCY PROVIDED BY SLS BATTERY SERIES ACTIVE REDUNDANT SLS CRUISE COMMUTATOR DATA SWITCHES & DRIVERS PROVIDE ACTIVE REDUNDANT DC-DC CONVERTER REGULATORS STANDBY REDUNDANT CBS CRUISE ENCODER SERIES ACTIVE REDUNDANT CBS CRUISE COMMUTATOR DATA SWITCHES & DRIVERS STANDBY REDUNDANT CBS CRUISE COMMUTATOR DATA SWITCHES & DRIVERS STANDBY REDUNDANT BATTERY FLOAT CHARGERS REDUNDANT REFING CUTTERS FOR EACH PARACHUTE REEFING LINE (1 OF 3 REQUIRED) MULTI AXIS GYRO SENSING. ACTIVE REDUNDANT CRUISE COMMUTATOR RELAYS ACTIVE REDUNDANT COMMAND DECODER RELAYS ACTIVE REDUNDANT COMMAND DECODER RELAYS SERIES ACTIVE REDUNDANT CBS CRUISE MONITOR CONTROL DATA SWITCHES & DRIVERS FOUR LANDING RADAR VELOCITY SENSOR CHANNELS, (3 REQUIRED) REDUNDANT ADAPTOR CRUISE COMMUTATOR AND CRUISE ENCODER ONE OF TWO G AND C COMPUTERS SELECTED DURING IN-FLIGHT CHECKOUT STANDBY REDUNDANT BATTERY FLOAT CHARGERS ACTIVE REDUNDANT BATTERY CHARGER RELAY ACTIVE REDUNDANT		
	2147 2258 2602 2646 2919 2960 3034	.9323 .9328 .9340 .9345 .9361 .9374 .9379	94.61 95.47 97.39 98.43 101.30 104.28 105.19	.9617 .9623 .9634 .9635 .9636 .9650 .9651	.9688 .9700 .9715 .9758 .9772 .9784	.8354 .8364 .8364 .8364 .8376	TRIPLE REDUNDANT FREQUENCY DIVIDERS WITH MAJORITY VOTERS DUAL CARTRIDGE EXPLOSIVE BOLTS – PARACHUTE RELEASE ACTIVE REDUNDANT ESP SCIENCE DATA REMOTE :NTERFACE ELECTRONICS STANDBY REDUNDANT ESP COMMUTATOR AND ENCODER SERIES REDUNDANT PRESSURE REGULATOR (RCS) STANDBY REDUNDANT ESP PROGRAMMER		
	3096 3430 3493 3572 3966 4240 4663	.9395 .9396 .9409 .9450 .9455 .9456	107.07 109.02 109.15 112.78 123.86 125.34 125.69 134.56	.9668 .9668 .9671 .9713 .9718 .9718	.9792 .9801 .9802 .9802 .9844 .9850 .9853	.8384 .8391 .8391 .8441 .8478 .8483 .8483	ACTIVE REDUNDANT TRANSMITTER TUBES IN RADAR ALTIMETER PROVIDE ACTIVE REDUNDANT RESISTANCE HEATERS INCORPORATE TRIPLE REDUNDANT DECREMENTERS AND ZERO DETECTORS STANDBY REDUNDANT SLS TV DATA PROCESSER VELOCITY AND RANGE SENSOR REDUNDANCIES IN LANDING RADAR ACTIVE REDUNDANT DISCRETE OUTPUT LINE DRIVERS ACTIVE REDUNDANT THERMOSTATS STANDBY REDUNDANT SLS COMMUTATOR AND ENCODER		

s not include experiments.

des a weight factor.

weement of a Flight Capsule Landing.

(5) Performance of Entry Experiments.(6) Performance of Landed Experiments.(7) Incorporated Redundancy.

are in the Capsule Bus System. The telecommunication redundancies improve the estimated reliability of commutating and encoding engineering data, and in addition interleave the post separation engineering data with the ESP telecommunications. The redundancies in the CB electrical power system improve the reception of regulated power from the spacecraft and provide CB battery backup by the SLS Battery. Three guidance and control redundancies improve the estimated reliability of the radar altimeter and landing radar. Dual cartridges for separation devices, dual initiators and multiple reefing cutters were provided by the staging redundancies. The resulting reliability for achieving successful landing is 87%.

Addition of redundant components increases both vehicle weight and reliability, as illustrated in Figure 4.10-5. Point A on this figure represents a design without redundancy and point B a design including the six items necessary to meet the constraints. Our preferred design, with a reliability of all equipment of .71, is represented by point C. The potential reliability if all 45 of the recommended redundancies could be incorporated is indicated by point D. The line from A to B to D represents the maximum reliability available when the design includes the six constraint-required items.

The parallel paths represented by use of two Planetary Vehicles improves the probability of successful landing. For our preferred design, the reliability of all equipment for performing successful entry experiments with at least one of the two capsules is 98% and successful landing to 98%. All of these estimates are conditional upon the successful operation of the Flight Spacecraft, the Launch Vehicle, and other VOYAGER systems which support the Flight Capsule.

RELIABILITY IMPROVEMENT THRU EFFECTIVENESS ANALYSIS

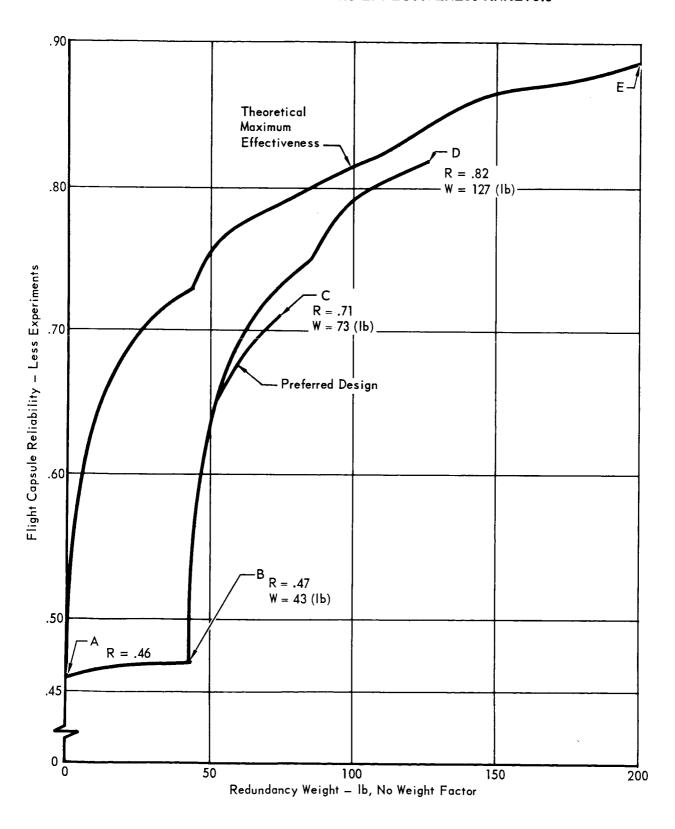


Figure 4.10-5

SECTION 4

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